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High Reynolds Number Hybrid Laminar Flow Control (HLFC) Flight Experiment III. Leading Edge Design, Fabrication, and Installation

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PREFACE

The program was jointly sponsored by NASA; the United States Air Force, Wright Laboratory, Flight Dynamics Directorate; and The Boeing Company. The contract was managed by Mr. R. D. Wagner, Head of Laminar Flow Control Project Office, and Mr. D. V. Maddalon, Technical Monitor. Mr. R. L. Clark was the Wright Laboratory (WL/FIMM, Wright-Patterson Air Force Base, OH) Program Manager. The period of performance was from December 1987 through August 1991.

The program was conducted by the Advanced Development Aerodynamics organization of Boeing Commercial Airplane Group (BCAG), supported by the BCAG Nacelle, Strut, and Propulsion System Engineering, BCAG Structures Engineering, BCAG Mechanical/Electrical Systems Engineering, and BCAG Flight Test organizations.

The principal contributors to the work described here are Mr. J. W. Fogleman, Mr. K. W. Johnson, Mr. J. W. Williams, and Mr. D. L. Grande of BCAG Structures Engineering; Mr. T. A, Munns of Boeing Materials Technology; and Mr. D. H. Gane of BCAG Manufacturing Research and Development. Mr. F. J. Davenport of BCAG Aerodynamics Technology and Mr. F. M. DaRonch of BCAG Structures Engineering provided technical and document integration services. Mr. A. L. Nagel of BCAG Aerodynamics Technology, HLFC program manager, gave guidance and suggestions throughout the course of design effort.

Special thanks are owed to Mr. R. D. Wagner and Mr. D. V. Maddalon of NASA Langley Research Center, who generously contributed time and effort to make their unique backgrounds of laminar flow expertise and flight test experience available to Boeing personnel.

Special recognition must be given to Mr. B. Humpheys and his associates at TKS (Aircraft Deicing) Ltd. for their dedication to the development of the process for perforating the titanium panels used for the leading-edge skin. Without their efforts, the probability of success would have been far less.

Finally, the authors wish to acknowledge the vital contribution of Dr. Werner Pfenninger. In addition to being a mentor to all participants in modern laminar flow control work, his broad knowledge of the structural arrangement and details required to achieve laminar flow significantly aided the success of the HLFC program

SYMBOLS AND ABBREVIATIONS

BAJ	Bond assembly jig
EB	Electron beam
HLFC	Hybrid laminar flow control
LE	Leading edge
LEFT	Leading edge flight test
M_{mo}	Maximum operating Mach number
OSS	Outboard slat station
TAI	Thermal anti-icing
V_{mo}	Maximum operating indicated airspeed
WBL	Wing buttock line
WRP	Wing reference plane

1.0 SUMMARY

A Hybrid Laminar Flow Control (HLFC) suction panel was designed, fabricated, and installed on a Boeing 757 airplane, along with the required support structure, ducts, and valves. The panel and suction system were developed to permit flight demonstration of HLFC at high Reynolds number on a modern turbofan-powered transport and to conduct flight research on laminar flow control technology. Two Krueger flaps replaced the two production airplane slats on the leading edge of the HLFC panel. The integrated HLFC test article met the aerodynamic smoothness requirements established in volume II of this report and provided the necessary ducting to apply the appropriate suction to the panel surface. The test article also met all additional design criteria established for the HLFC experiment, including–

a. Compatibility with existing aircraft structure.

b. Component thermal expansion compatibility.

c. Strength for aerodynamic maneuver loads.

d. Maintenance safety level equivalent to that of a production airplane.

The suction surface was constructed using state-of-the-art laser perforating technology that provided the nonuniform perforation density required to permit the tailoring of local suction rates. Hot forming and bonding processes were developed to meet the high quality requirements imposed on the finished panel.

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2.0 INTRODUCTION

2.1 BACKGROUND

The potential for reducing wing friction drag by increasing the extent of laminar flow was recognized more than half a century ago. However, boundary layer instabilities associated with high Reynolds number and with sweepback prevented achievement of significant laminar runs on the wings of large high-performance airplanes. In the 1960s, the USAF X-21 program (ref. 1) showed that those problems could be overcome by using slot suction to stabilize the boundary layer, provided that care was taken to control wing surface roughness and waviness. The program failed as a demonstration of practical laminar flow control because of a flawed joint design that required continual repair or replacement of aerodynamic smoothing material. There was also debate as to whether the complexity of a suction system that covered the entire wing with slots and subsurface ducts was justified by the performance gain.

The concept of hybrid laminar flow control (HLFC), invented by L. B. Gratzer of The Boeing Company (U.S. Patent No. 4,575,030), greatly simplifies laminar flow control by confining suction surfaces and pneumatic system components to the leading edge. HLFC maintains laminar flow downstream of the wing front spar solely by tailoring the pressure distribution.

Other concerns relating to anti-icing and to clogging or roughening of suction surfaces due to insect accretion were addressed by the NASA Leading Edge Flight Test Program (refs. 2, 3, and 4). A modified Lockheed Jetstar airplane equipped with a partial-span leading-edge suction system was flown in a variety of hostile environments and demonstrated reliable operation.

The present program was sponsored by NASA, with partial USAF sponsorship and Boeing participation, in order to—

a. Perform high Reynolds number flight research on HLFC.

b. Obtain data on the effectiveness of HLFC on a large high-subsonic-speed transport airplane.

c. Develop and demonstrate practical design concepts for HLFC systems.

2.2 TECHNICAL APPROACH

A Boeing-owned 757 airplane was modified to include all the critical systems for a full-scale HLFC application, plus flight-operable suction controls and extensive instrumentation to meet HLFC research requirements. The 757 was ideally suited for the program because its advanced aerodynamic technology wing permitted attainment of the needed HLFC pressure distribution with only a small contour change ahead of the front spar; and the smoothness of the existing between-spar structure allowed the test to be conducted with minimal fairing or coating beyond normal paint. This ensured that the data obtained would have practical application to standard production wings and not be restricted to ideally smooth surfaces.

2.3 PROGRAM TASKS

The program effort consisted of the following categories:

a. **Aerodynamic Design**. Definition of the surface pressures and suction levels required to achieve extended laminar flow, followed by geometric design of the wing contours needed to obtain the surface pressures. This task is treated in detail in volume II.

b. **Leading-Edge Structural Design and Fabrication**. The design, construction, and installation of a 22-ft section of wing leading edge having provisions for suction through a porous outer skin and for a Krueger-type leading-edge flap serving both as an integral part of the airplane high-lift system and as a shield against insect accretion at low altitude. The leading edge was required to meet stringent aerodynamic smoothness and waviness requirements under load and was also required to provide structural integrity. This task is treated in detail in volume III (this volume).

c. **Suction System Design and Manufacture**. The design of the system of air passages, ducts, valves, and pump, and the specification of leading-edge outer skin porosities. The system was required not only to provide the suction flows required for laminarization but also to demonstrate anti-icing capability. To achieve this, hot pressurized air was required to flow out through certain porous portions of the skin. The system was also required to provide for a wide range of suction flow adjustment to permit optimization of HLFC suction quantities and to permit generation of boundary layer behavior data under a variety of suction conditions, in support of research on boundary layer analysis methods. This task is reported in detail in volume IV.

d. **Flight Test and Data Analysis**. The definition and installation of suitable instrumentation to evaluate boundary layer conditions and suction system performance, followed by the conduct of the tests, data acquisition, and evaluation of test results. This task is reported in volume I, along with an overview of the program as a whole.

3.0 REQUIREMENTS

3.1 GENERAL

The HLFC test article consisted of a boundary layer suction system installed in a modified leading edge and engine nacelle strut on the left outboard wing of a Boeing 757 airplane, as shown in figure 3.1-1.

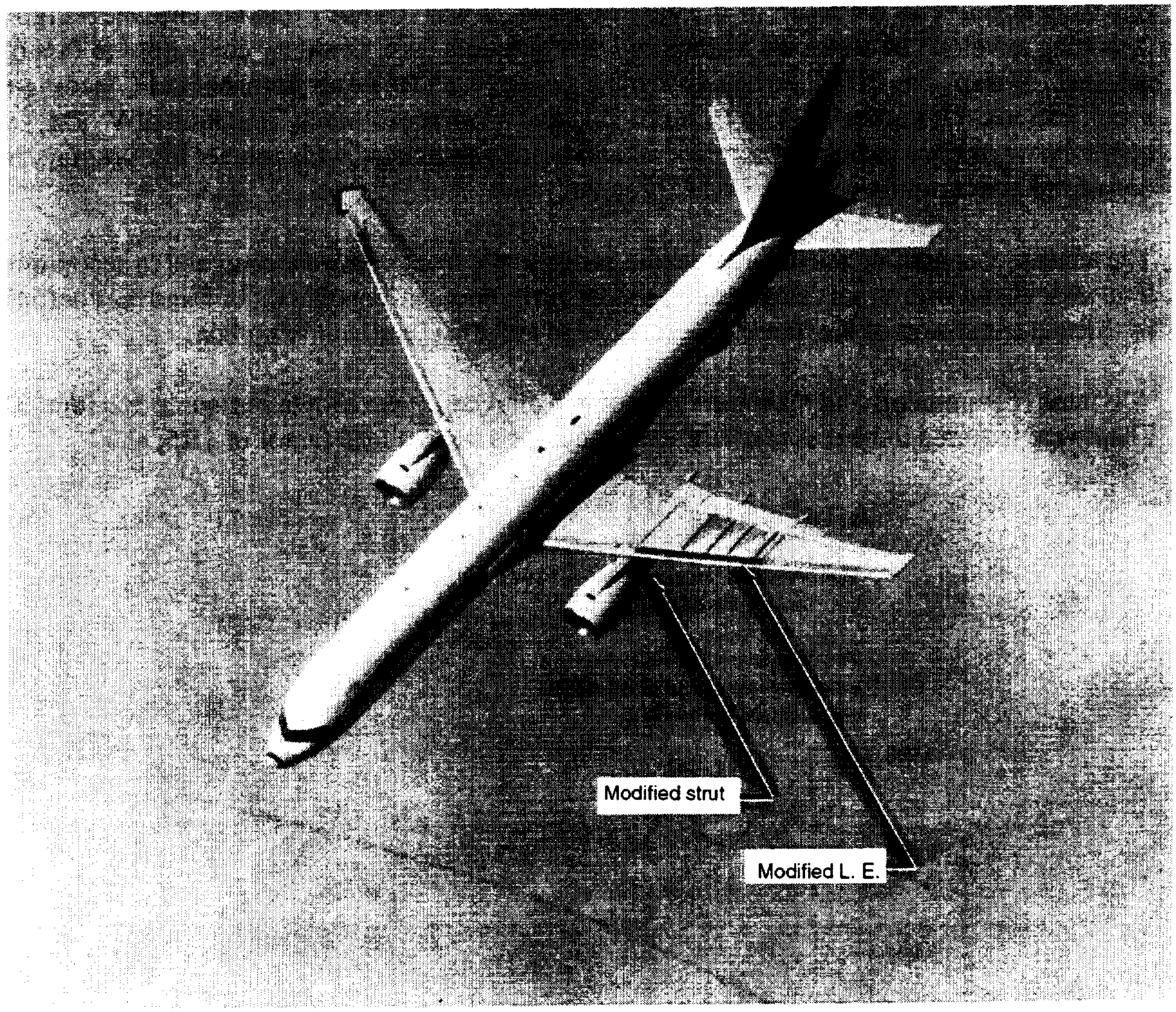

Figure 3.1-1. 757 Airplane Modified for HLFC Flight Test Program

The 22-ft wing modification replaced two production airplane slats and the underlying fixed leading edge with a suction panel, valving, ducting, and two Krueger flaps. The skin over the outboard 17 ft had perforations and was provided with suction ducts for the boundary layer suction experiment.

The outer skin surface and spar joint were the most critical manufacturing details of the entire design. They were required to be smooth and free of waves to a much higher standard than conventional airplanes. Great pains were taken with the design, manufacture, and installation of the test panel and the spar joint to ensure that these standards were met.

The nacelle strut modification included fairings shaped to accommodate the turbocompressor, heat exchanger, control valves, and other system components.

3.2 STRUCTURAL DESIGN

All the new structure and structural modifications were designed to meet the same design requirements applied to the original 757 airplane. Likewise, if the margins of safety on unmodified structure had been negative as the result of the modifications, it was intended that the airplane weight and operating envelope be restricted as required to maintain a level of safety equivalent to that of the original 757 airplane. In fact, no negative margins were found to result from the HLFC leading-edge modifications. (A maximum load factor of 2.0g and an airspeed/Mach limitation of V_{mo}/M_{mo} were applied nevertheless because of airplane modifications associated with another test program being conducted concurrently with the HLFC program.)

The attachments holding the titanium leading edge to the front spar were designed to accommodate loads associated with strain compatibility and the differences in Young's moduli and the coefficients of thermal expansion of the titanium leading edge and the aluminum wing box.

Waviness requirements and maximum step or gap allowables are shown in figures 3.2-1and 3.2-2. They were required to be met at cruise load conditions up to a load factor of 1.15.

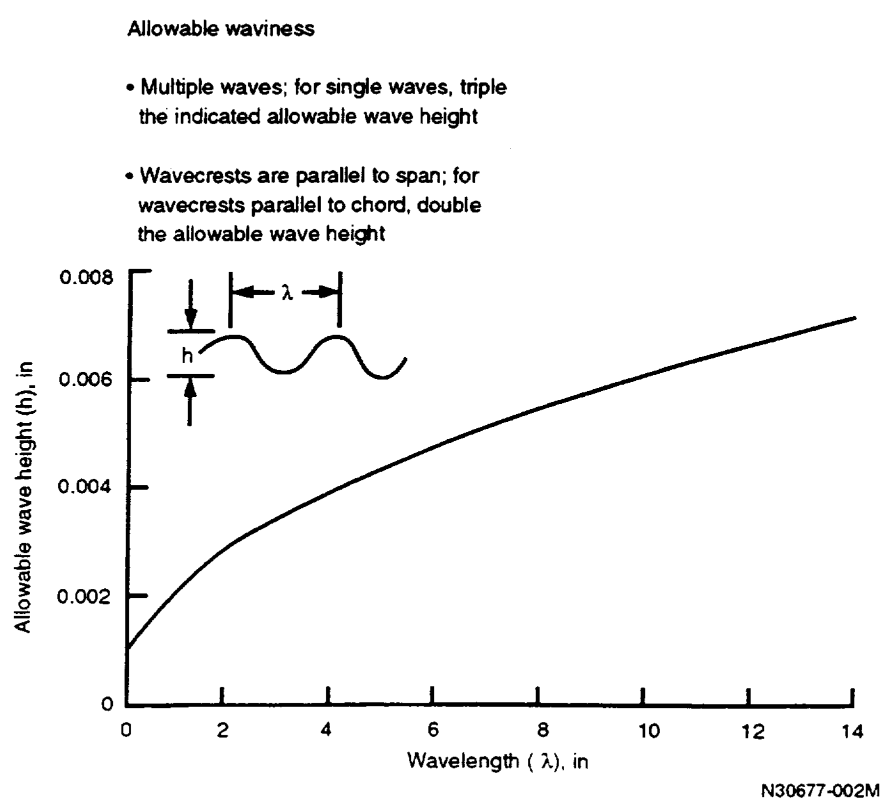

Figure 3.2-1. Allowable Waviness

Spar-Joint Step or Gap—Maximum Allowable Discontinuity

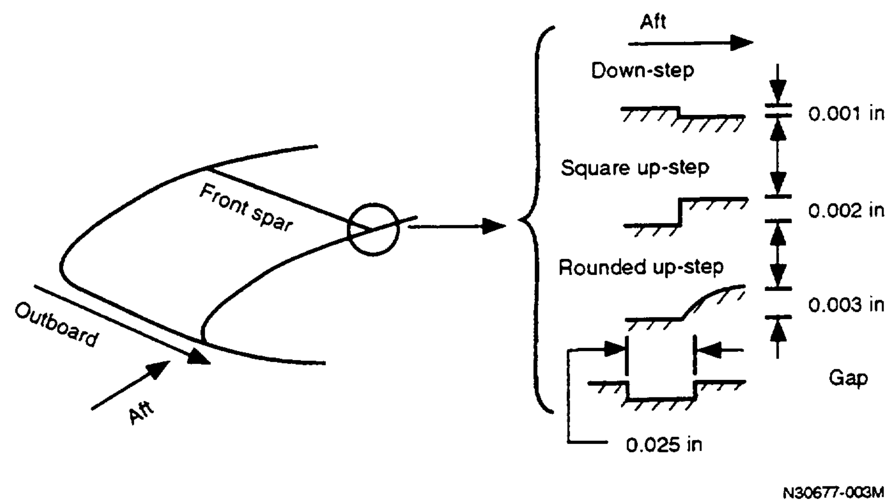

Figure 3.2-2. Panel and Joint—Step or Gap Criteria

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4.0 LEADING EDGE

4.1 GENERAL DESCRIPTION

The HLFC leading-edge structure replaced two slats, their support structure and drive mechanism, and the fixed leading-edge structure beneath the slats.

The new leading-edge structure consisted of an upper surface suction panel, a substructure of leading-edge ribs, two Krueger flaps and drive linkages, and a small fixed lower panel between the flap and the front spar. Figure 4.1-1 is a photograph of the upper surface with the leading edge installed and Krueger flaps extended.

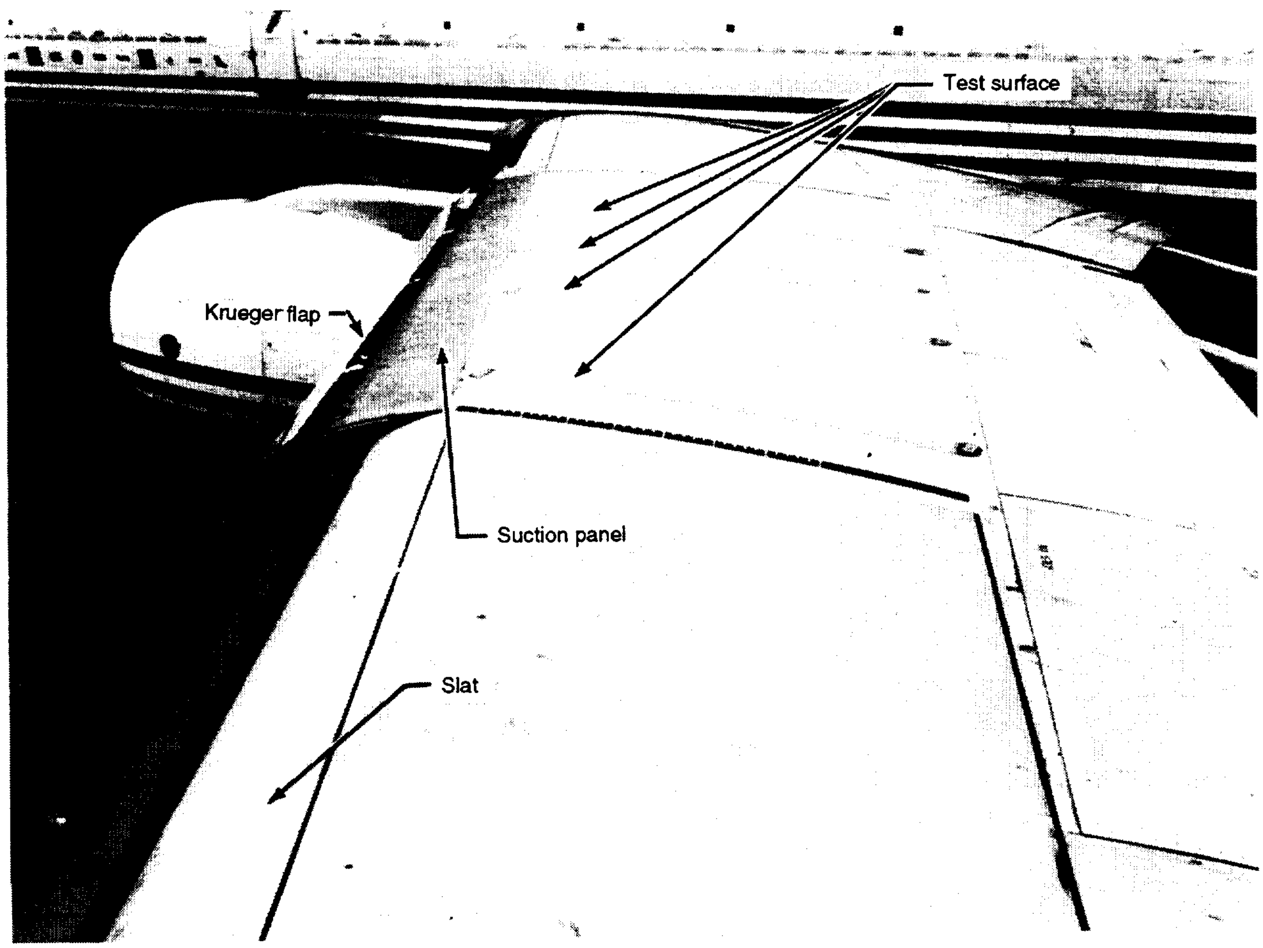

Figure 4.1-1. Leading Edge Installed on Wing (Krueger Flaps Deployed)

The suction panel was an adhesively bonded titanium structure consisting of two skins: the outer perforated and the inner solid. These were spaced apart by spanwise inner member stringers that provided panel surface stiffness and at the same time formed discrete air transport bays. Additional details of the panel are given in section 4.3.

The ribs, aluminum hogouts with titanium chords, supported the panel spanwise at twenty-one locations: one support at each end, two at each of four Krueger support points, and eleven at intermediate positions. Figure 4.1-2 shows the underside of the suction panel and the extended Krueger leading-edge flaps. These flaps were a conventional design, with slaved folding bullnoses. They served both as high-lift devices and as shields against insect accretion on the suction surface in low-altitude flight.

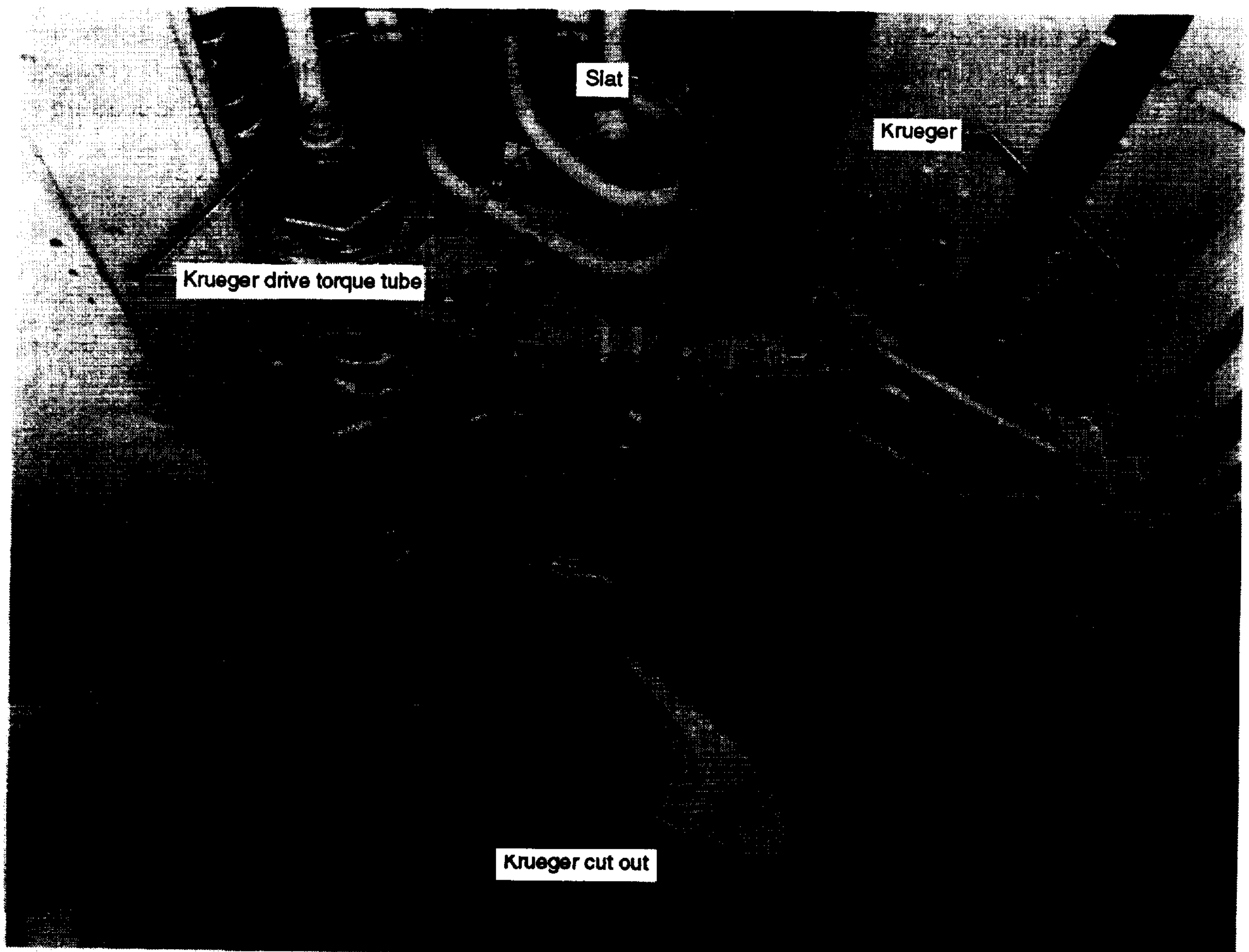

Figure 4.1-2. Leading Edge and Krueger Flaps (Looking Outboard)

4.2 SUCTION PANEL REQUIREMENTS

The suction panel was designed to meet the following requirements:

a. Aerodynamic surface roughness and waviness criteria (sec. 3.2, above).

b. Outer skin perforation diameters and spacings, as shown in figure 4.2-1. (This pattern was determined as described in vol. IV to meet the suction flow requirements determined in vol. II.)

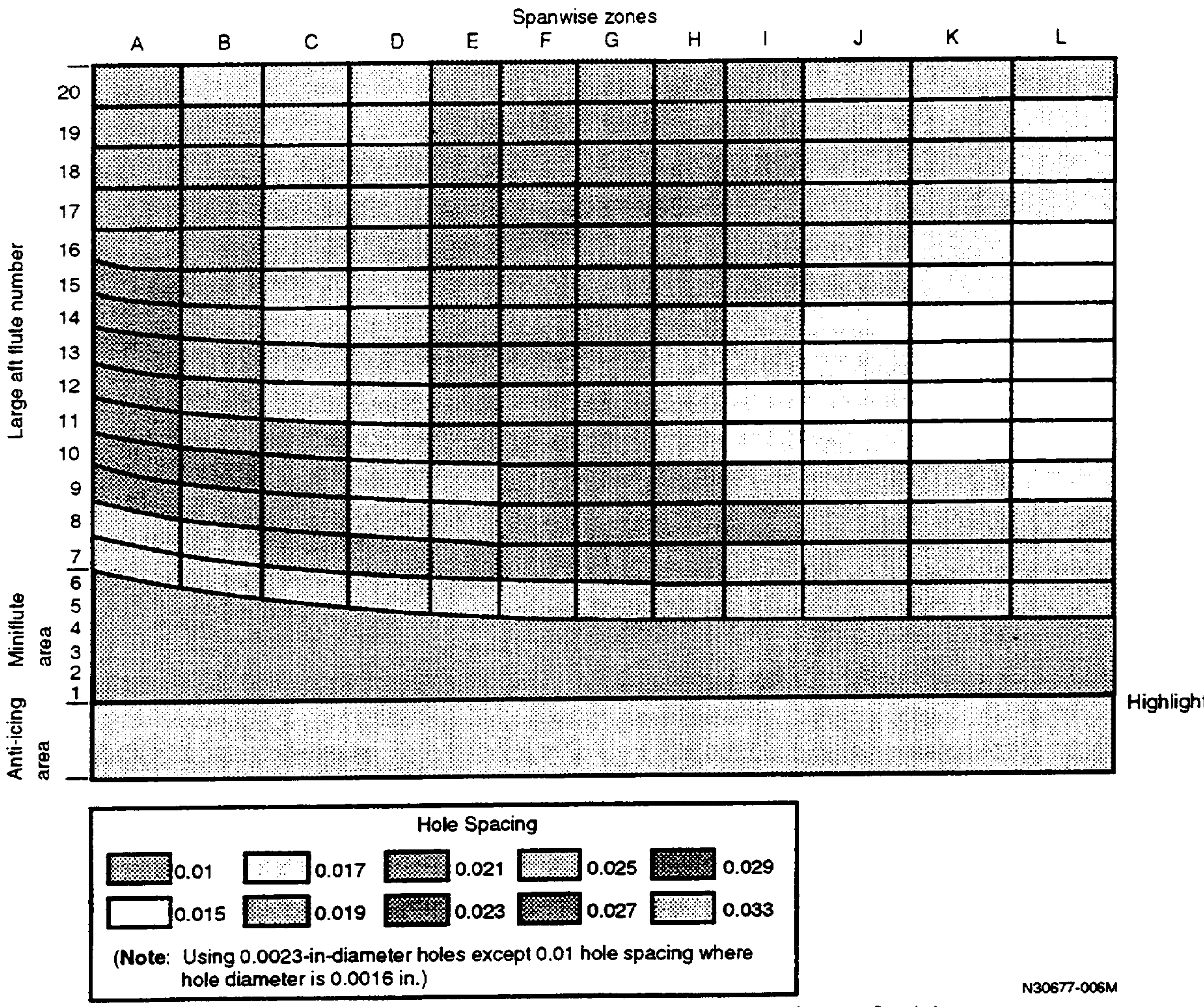

Figure 4.2-1. Outer Skin Perforation Pattern (Not to Scale)

c. Location and spacing of stringers between the inner and outer skins to form spanwise passages ("flutes") for the suction airflow. The size and location of the flutes are critical to the operation of the HLFC system and they impose the following constraints on stringer geometry:

1. The flute cross sections must be large enough to accommodate the required suction flow.

2. At least 50% of the outer skin must be available for boundary-layer suction. Therefore, the stringer spacing must be at least equal to the stringer width.

3. In the forward 4 to 6 inches of the suction panel, the stringers must follow the design-condition isobars of the external flow. These isobars curve forward with increasing distance outboard, as shown in figure 4.2-2.

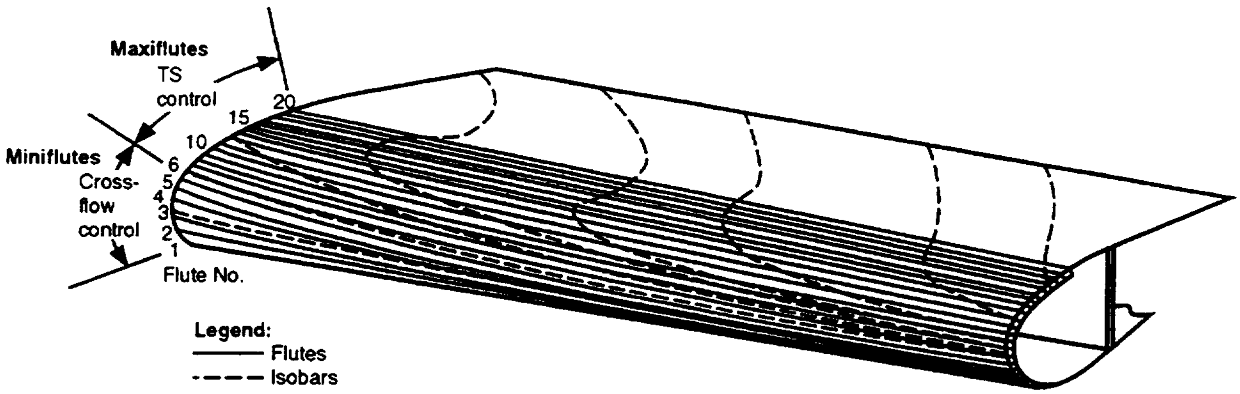

Figure 4.2-2. Flute Layout and Isobar Location (Not to Scale)

4. The flutes must be narrow enough that the external pressure gradient will not result in a requirement for excessively low flute pressure to prevent outflow. This constraint is most severe near the nose, where the spacing is limited to 0.3 in.

d. Thermal stress due to dissimilar materials. The panel is subjected to thermal stresses induced by temperatures ranging from -65° to +310°F when the hot-air thermal anti-icing system is turned on, sending hot air to a cold-soaked wing.

e. Induced loading from wing-box bending. The panel must withstand the loads induced by wing-box bending while meeting the smoothness and waviness requirements stated above at an HLFC operational load factor of 1.15 and it must provide strength for a limit load factor of 2.5.

f. Aerodynamic loading. Panel deflections under aerodynamic loads, with an internal leading-edge cavity (the volume in front of the spar and between the suction panel and stowed Krueger flaps) overpressure of 1.333 psi, must not result in violation of the smoothness and waviness requirements.

g. Compatibility with the existing wing structure.

4.3 SUCTION PANEL DESIGN

The suction panel design evolved from a combination of external and internal aerodynamic requirements, the available space within the leading edge, the Krueger flap hinge positions, the substructure support interfaces, and the available manufacturing capabilities. A general view is shown in figure 4.3-1.

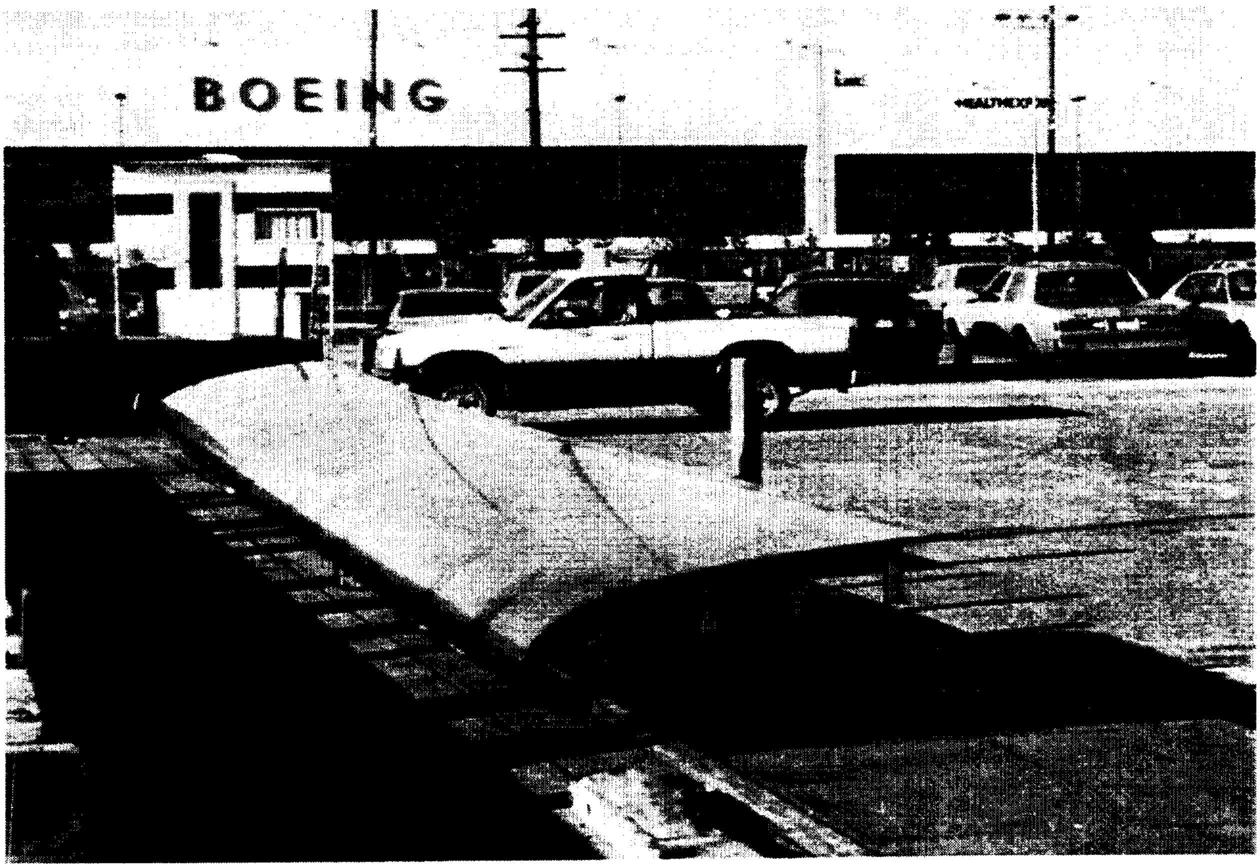

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Figure 4.3-1. Assembled Suction Panel

The panel operated as the initial airflow path for the suction system. Air was drawn from the external surface through the outer skin perforations and then directed spanwise through the flutes. It exited the flutes through metered orifices in the inner skin at four collector boxes and passed into a larger duct. A description of this flowpath is provided in section 4.0 of volume IV.

The new structure designed for the HLFC experiment was designed using "good aircraft practice." This standard provides for a minimum life of 1,500 hr without requiring tests.

The panel was 22.4 ft long, about 0.5 inch thick, and 25 inches from the leading edge to the front spar interface. It weighed 278 lb and had approximately 19 million holes in the outer skin. Titanium sheet of 0.040-in thickness was selected for the outer surface because it permitted laser-drilling the 0.0016- and 0.0023-in-diameter holes necessary for the suction system. Other components were also made of titanium for compatible thermal expansion.

There were three spanwise levels of structural complexity. The outboard suction zone had an inner and outer skin and stringer arrangement that provided suction passages. The inboard suction zone contained only a single outer skin and a reduced complement of stringers. A structural transition zone between the two provided for orderly discontinuation of the inner skin and some stringers.

The resulting design (figs. 4.3-2 and 4.3-3) consisted of a perforated titanium outer skin, machined (generally rectangular) titanium stringers, titanium end fittings, and a two-piece titanium inner skin adhesively bonded into a very stiff assembly.

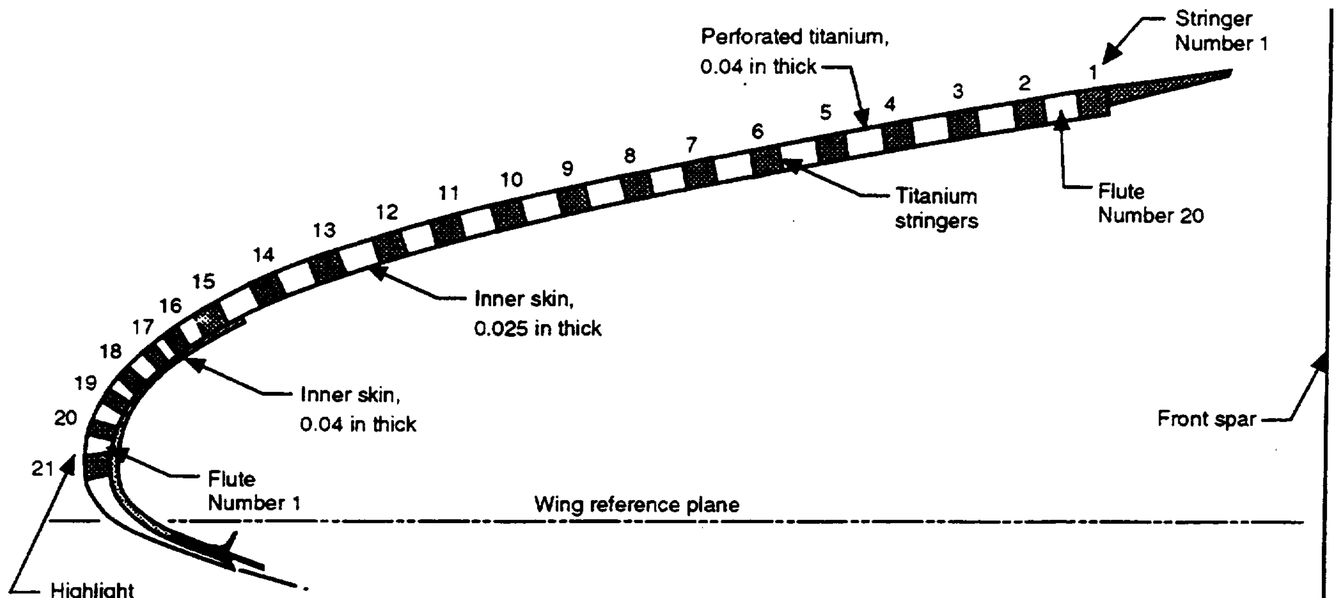

Figure 4.3-2. Suction Panel Cross Section

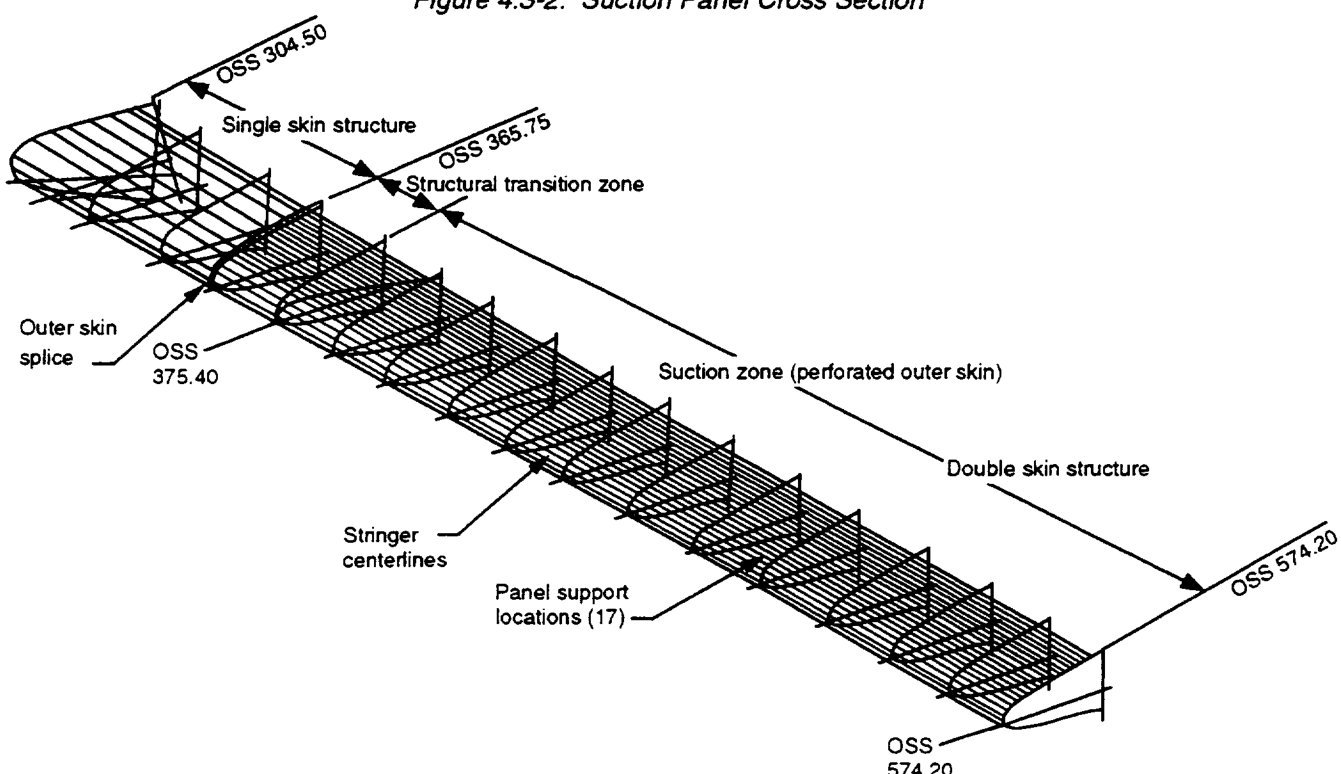

Figure 4.3-3. Panel and Stringer Outline

4.3.1 Stringers

From the highlight aft for 3 to 4 in. six narrow stringers, numbered 16 through 21 in figure 4.3-4, formed narrow passages called miniflutes. The stringers were curved to make the miniflutes run along lines of constant wing external pressure (isobars) at the design flight condition. They were 0.30 in wide and 0.42 in deep in cross section and were spaced approximately 0.30 in apart. Stringers 18 and 20 were machined into an "I" section and a channel, respectively, to provide increased flute area in order to reduce the spanwise airflow velocity. A view developed at the leading edge, illustrating the stringer runs, is shown in figure 4.3-5.

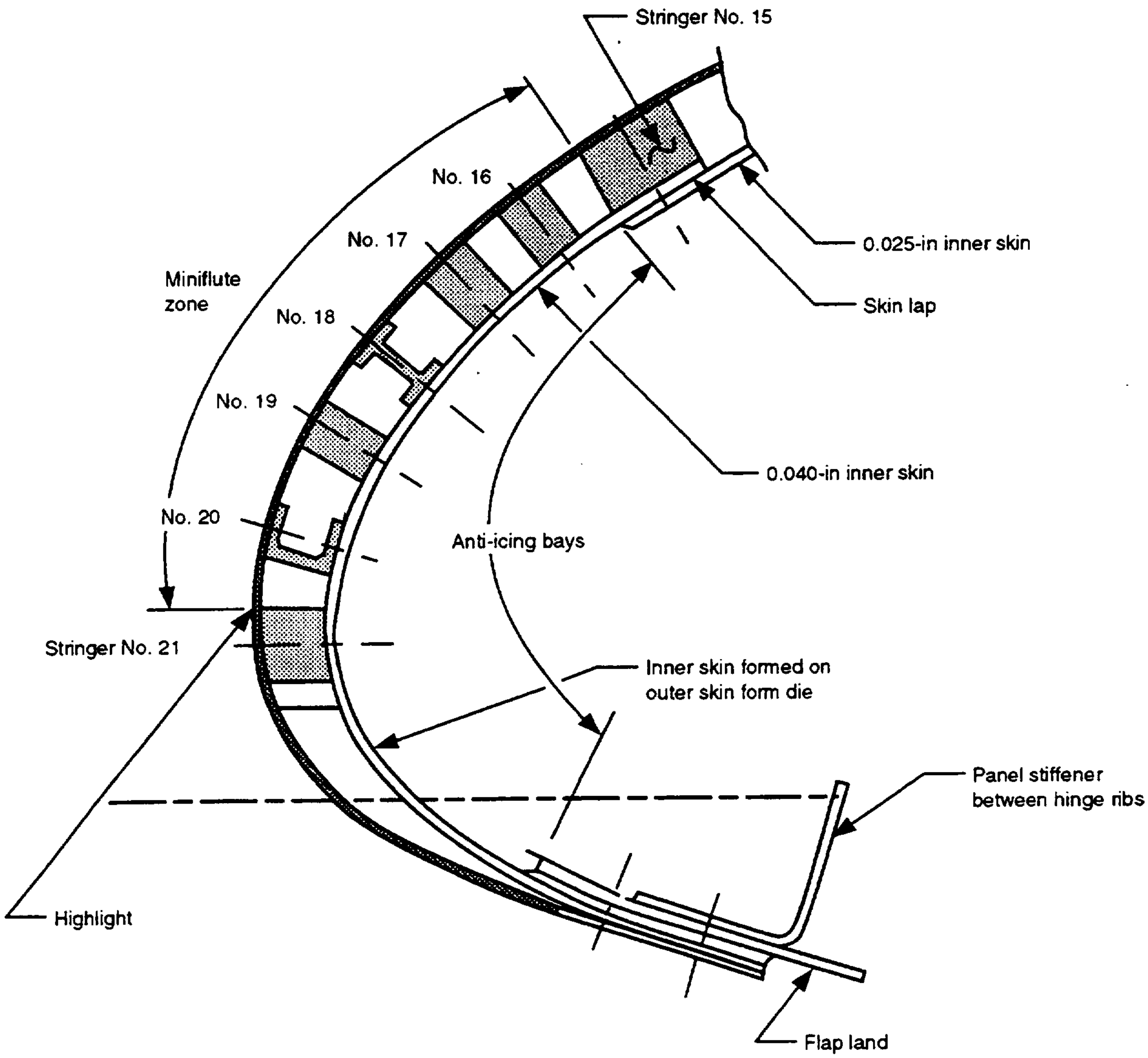

Figure 4.3-4. Miniflute Zone

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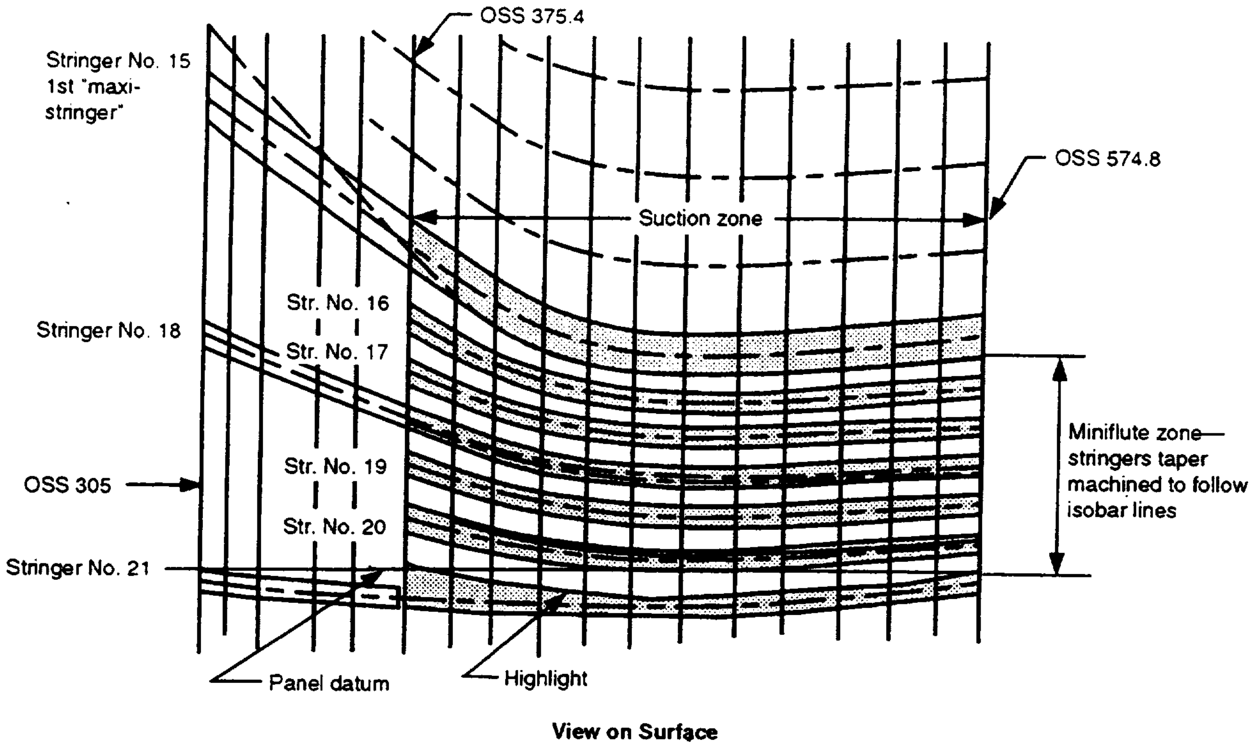

Figure 4.3-5. Stringer Runs in the Miniflute Zone N30677-012M

The remaining stringers, number 2 through 15, were 0.60 in wide by 0.40 in deep and were spaced approximately 1.2 in apart. Figure 4.3-6 shows local stringer details with rib attachment holes and edge chamfer for adhesive flow control. Threaded tooling holes were provided in some stringers for assembly purposes. These threaded holes did not extend into the skin. These were later plugged with threaded inserts to prevent inadvertent damage.

Stringer 1 (see fig. 4.3-6) provided the panel interface with the existing wing structure. Particular attention was paid to maintaining contour and slope at the panel trailing edge. The stringer was undercut to receive the aft edge of the perforated skin, and a radius conforming to the local wing contour was cut on the stringer outer surface. A radius was also machined on the inner surface to match and provide a nominal clearance with the faying interface on the main wing box. These details were crucial to the installation and are discussed more fully in section 4.7.

Stringers 1, 4, 7, 10, 13, 15, 18, and 21 were carried through to the inboard end of the panel. The remainder were tapered down inboard of OSS 375.4 to ease the transition loads, at this point, to the inboard unperforated panel.

Stringers were drilled and tapped at each rib station to receive the panel attachment bolts. Holes for instrumentation and assembly tooling were also drilled into the stringers. For this reason, and also because of the relationship of the outer skin perforation pattern to stringer position, close control of the final stringer position in the overall assembly was required. Therefore, a tolerance of ±0.01 in was applied to the stringer position at each rib station.

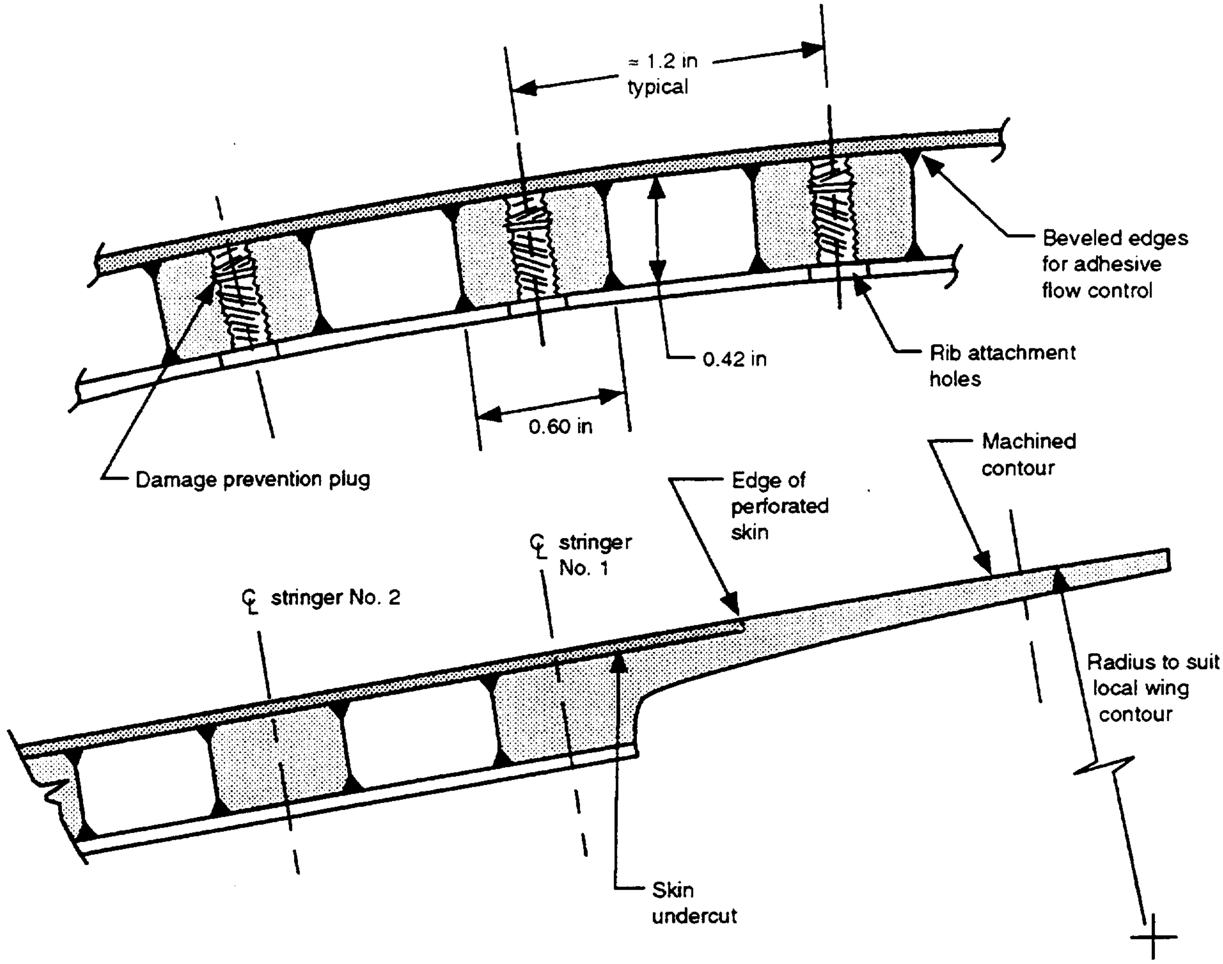

Figure 4.3-6. Stringer Details

4.3.2 Panel Details

Figures 4.3-7 and 4.3-8 show views of one of four cutouts in the panel required to clear the flap hinge fittings. The cutouts were designed to provide a working clearance with the extended hinge fitting, and at the same time, to be as far as possible from the aerodynamic attachment line. The cutouts precluded the use of a conventional flap land to provide a seal locally. Therefore, a spring tab was designed to travel with the flap and to seat against the lower surface of stringer 21 as it passed across the cutout. Figure 4.3-9 shows the operation of this tab.

4.3.3 Panel Skins

The relationship between the perforation pattern on the surface and the positions of the stringers, and the placement of both with respect to the aerodynamic profile, were critical. Therefore, a theoretical datum corresponding to the leading edge "highlight" was established, as shown in figure 4.3-10. This datum was used on the flat material to control and position the laser drilling of the perforation pattern, to control the position of the panel in the die during the forming cycle, and to set the final placement of the skin in the forming and bonding cycles. This permitted the final positions of the highlight, the

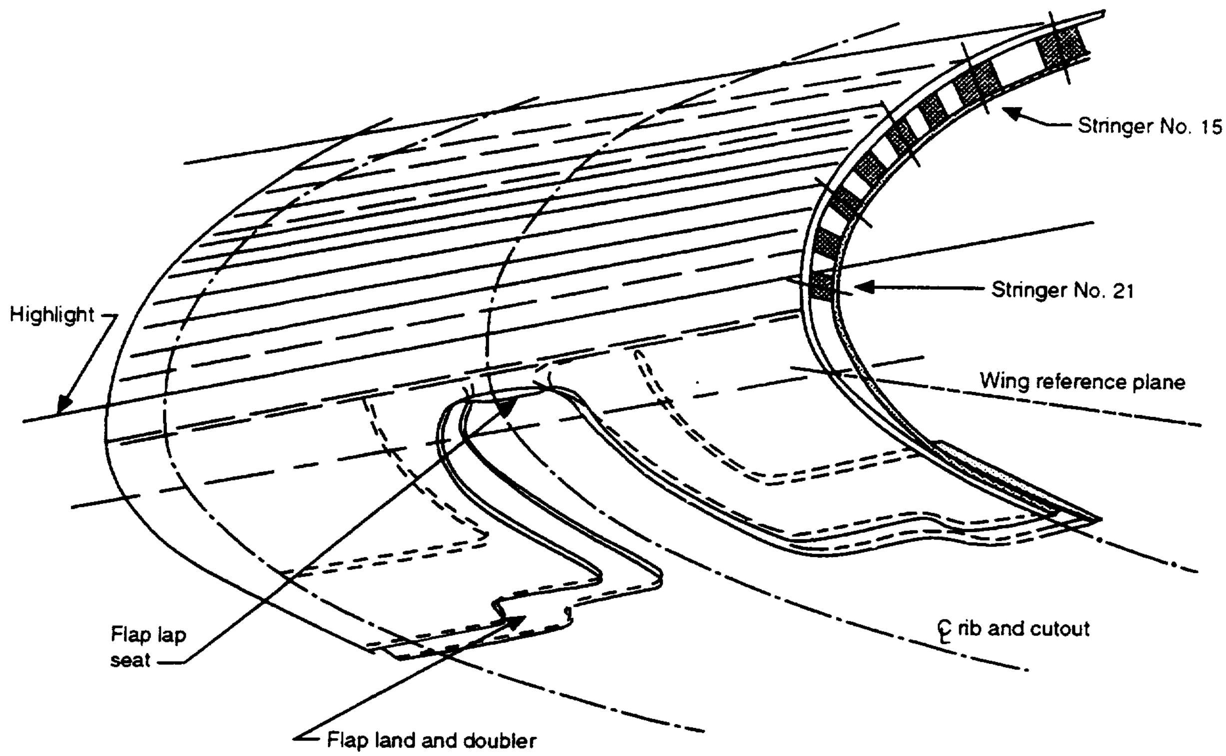

Figure 4.3-7. Hinge Rib Cutout View, Looking Up and Inboard N30677-014M

stringers, and the perforation pattern to be coordinated accurately and to meet the position tolerances required by the experiment.

A significant contour change occurred in the leading-edge cross section in the vicinity of the engine strut. To avoid possible forming complications in the critical test area, the external skin was divided into two pieces. The outer skin splice joint was located one bay inboard of the test section. The two external skins were easier to form than a single sheet during the hot forming cycle. The skins were connected by a bonded and riveted buttstrap joint during assembly.

The inner skin was made from commercially pure titanium sheet (CP-1). It was designed of two different thicknesses with an overlap joint under stringer 15 as shown in figures 4.3-3 and 4.3-4. The forward piece was 0.040 in thick, and the aft piece was 0.025 in. To eliminate additional tooling, both were formed in the same tool as the outer skin. The inner skins had holes for suction air extraction, panel attachment fasteners, and instrumentation porting.

4.4 PROCESS DEVELOPMENT AND FABRICATION

The suction panel fabrication and installation represented the greatest manufacturing challenge for the program because of the high-quality requirements imposed on the finished panel, the many processing developments that were driven by these requirements, and the manufacturing schedule.

The approach was to identify major process development requirements early and to complete development tests before fabrication of any flight hardware. The integration of all materials and processes was demonstrated on a full-scale, 4-ft segment of the suction panel. To demonstrate the

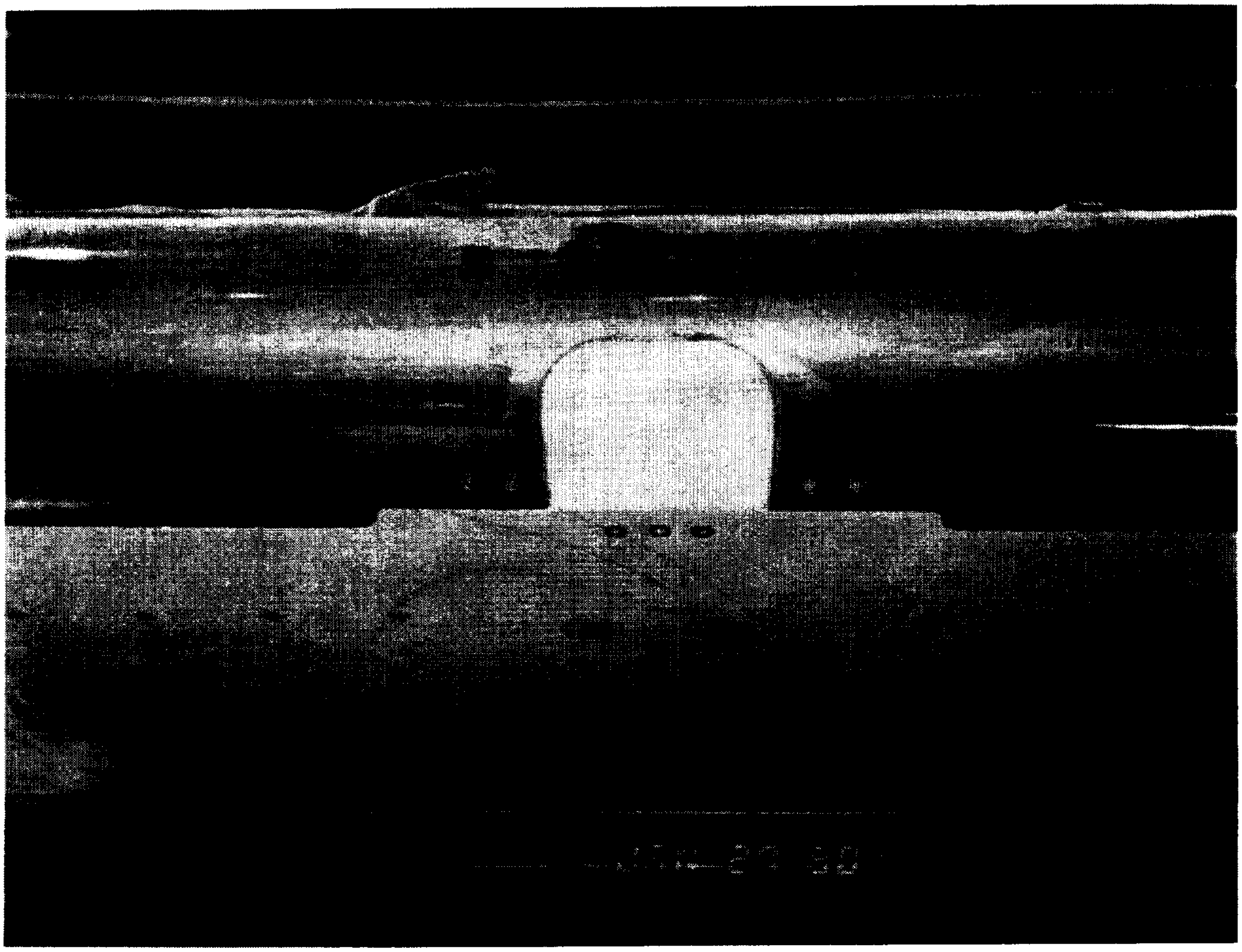

Figure 4.3-8. Leading Edge Showing Hinge Rib Cutout N30677-15M

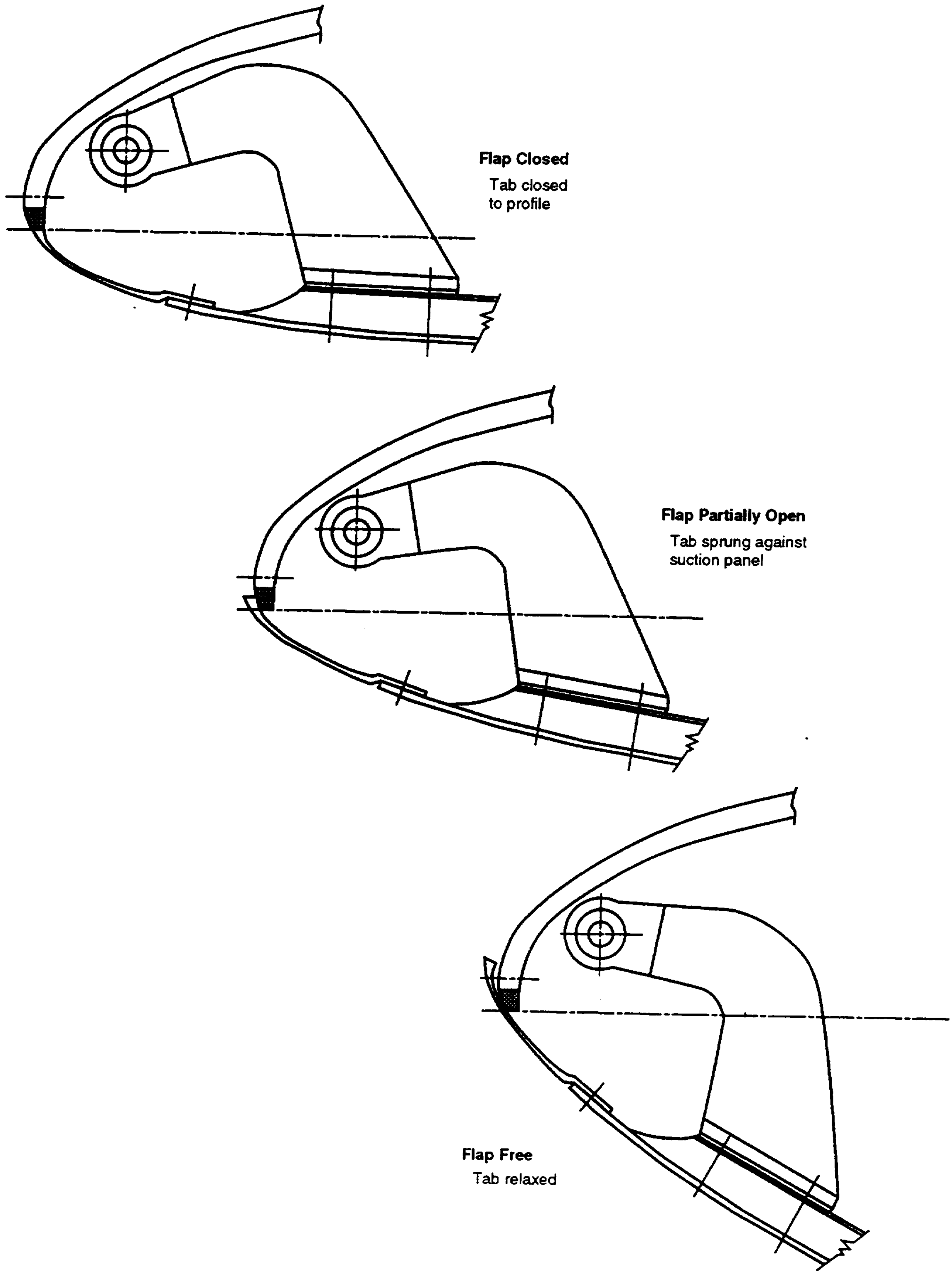

Figure 4.3-9. Suction Panel Development—Krueger Flap Seal at Highlight

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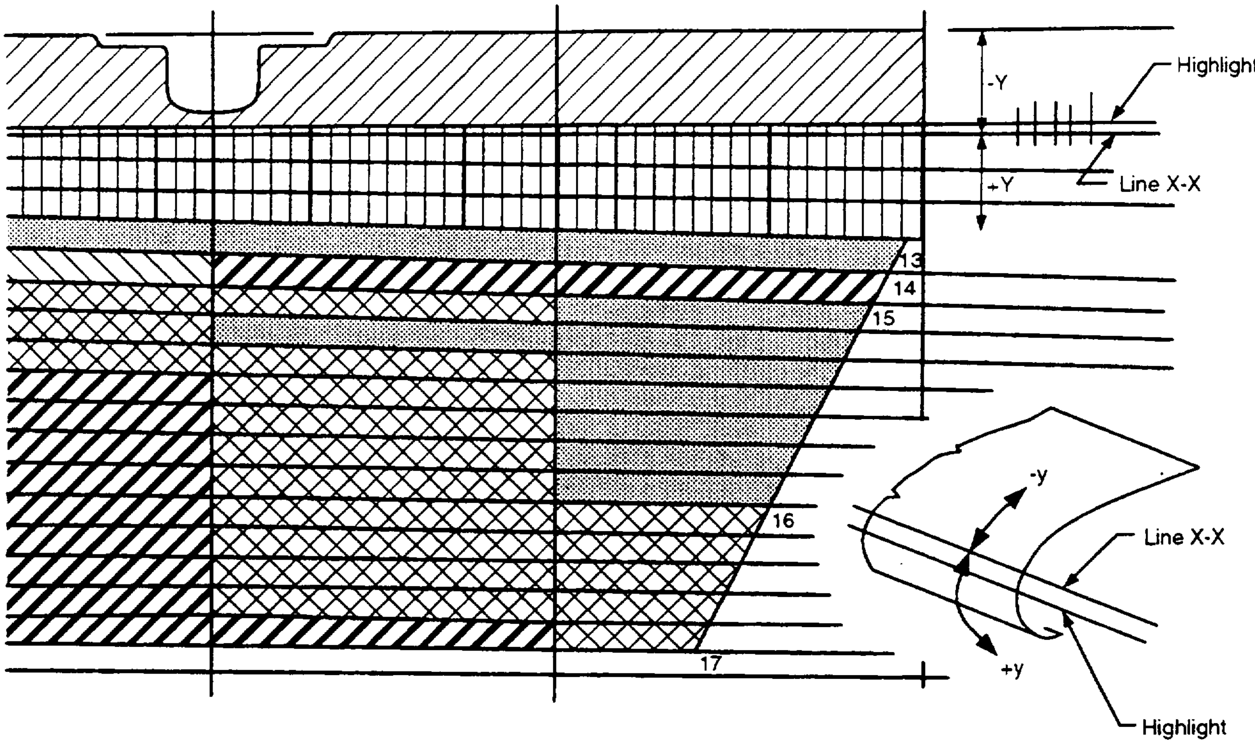

Figure 4.3-10. Suction Panel Datum Line and Relationship to Perforation Pattern

most difficult forming requirement, the segment was located at the inboard end of the panel (fig. 4.4-1) where the greatest change in cross section and the most severe double curvature were to be found. The finished verification panel is shown in figures 4.4-2 through 4.4-4.

After these processes had been demonstrated on the verification panel, fabrication of flight hardware commenced. During fabrication of the flight hardware, additional questions and risks were identified. Solutions to the significant processing risks were verified to the extent possible before processing began for the flight hardware.

The major process development areas included perforating and forming the outer skin, bonding the skin and stringers to the inner skin, and attaching the suction panel to the support ribs and wing box. The basic assembly sequence for the suction panel is shown in figure 4.4-5.

4.4.1 Perforation Process Development

Hole diameter and spacing were critical to the success of the HLFC concept. The state of the art in perforated titanium at the time of the HLFC flight experiment proposal was defined by the material used on the NASA Leading Edge Flight Test (LEFT) Program (ref. 4). That leading edge was built from material perforated by Pratt & Whitney using electron beam (EB) equipment. The size of a sheet that can be perforated using EB drilling is limited by the dimensions of the necessary vacuum chamber. Also, the process is restricted to thickness/diameter ratios of 10. Because hole diameters of about 0.0025 in were desired, the skin thickness could not exceed 0.025 in. This light gauge makes the leading edge susceptible to waviness due to imperfections in the substructure and to dents caused

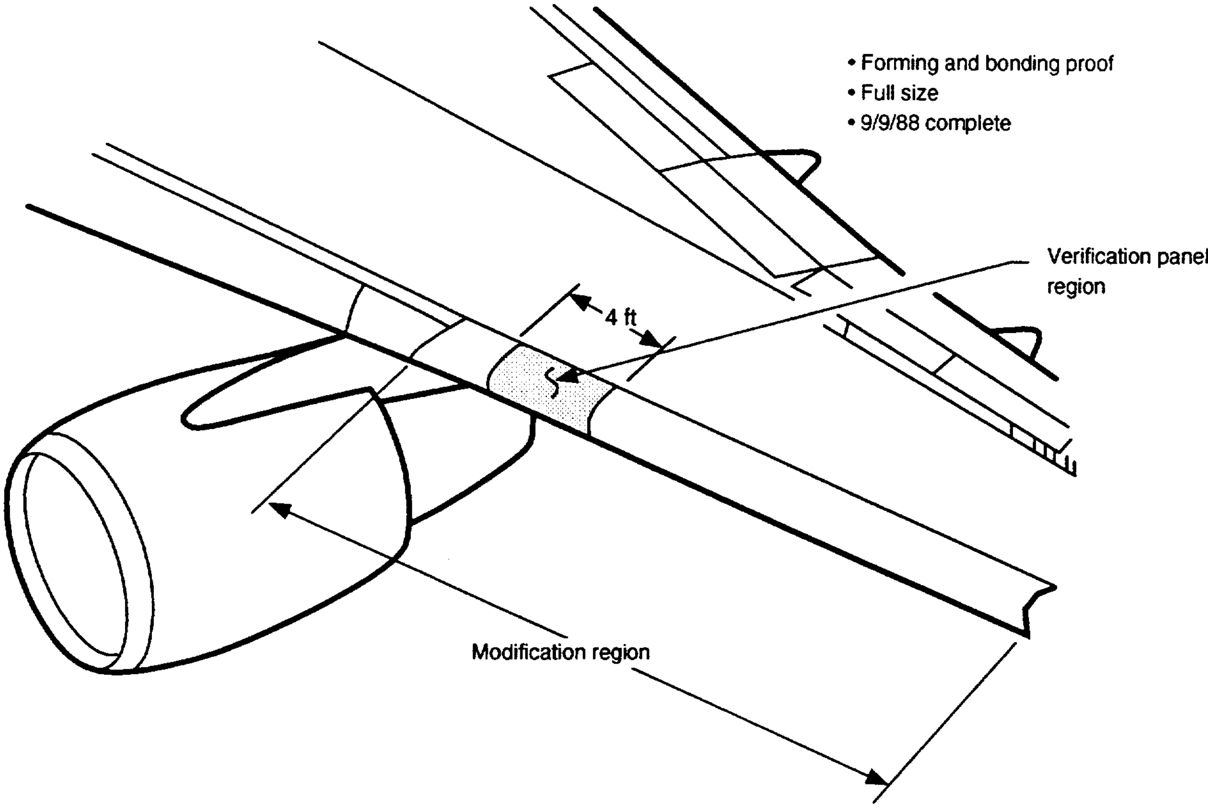

Figure 4.4-1. 4-ft Verification Panel

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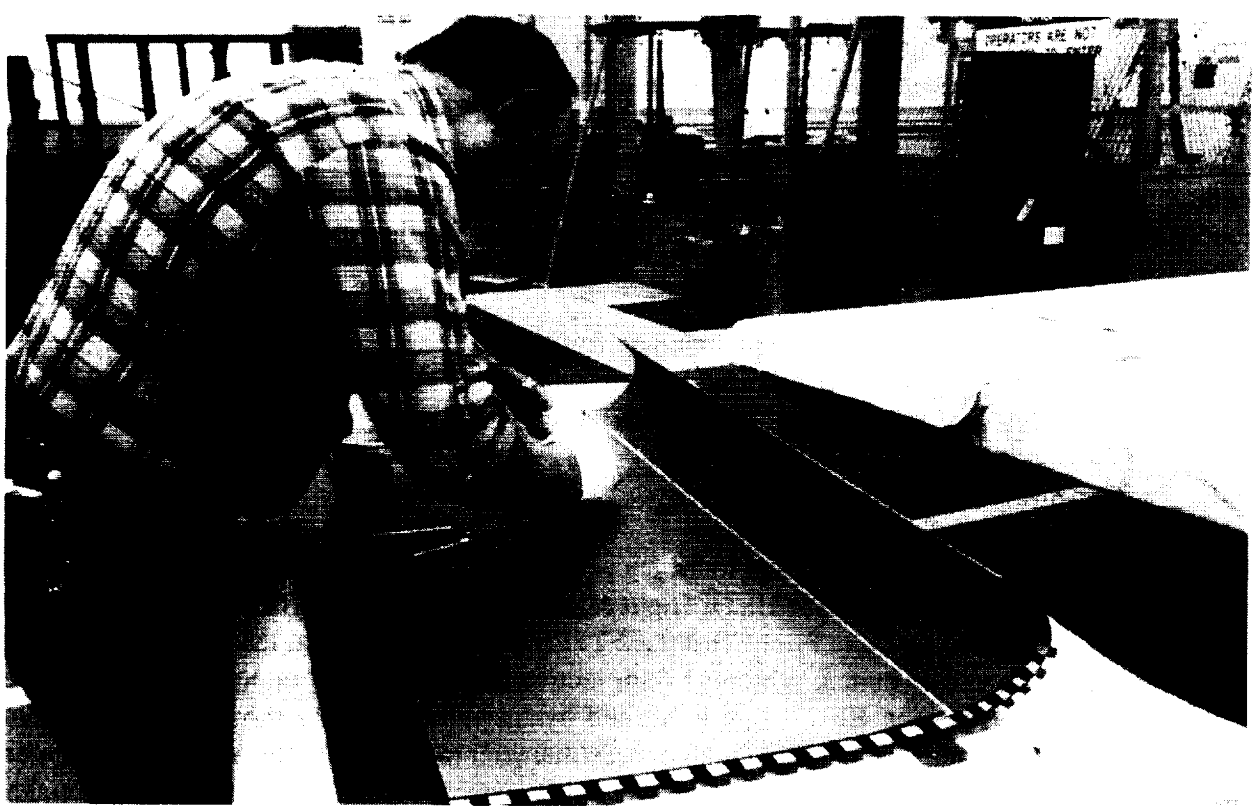

Figure 4.4-2. Stringer Assembly on Verification Panel

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N30677-20M

Figure 4.4-3. Checking Waviness of Verification Panel

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Figure 4.4-4. Checking Flow Rate Through Verification Panel

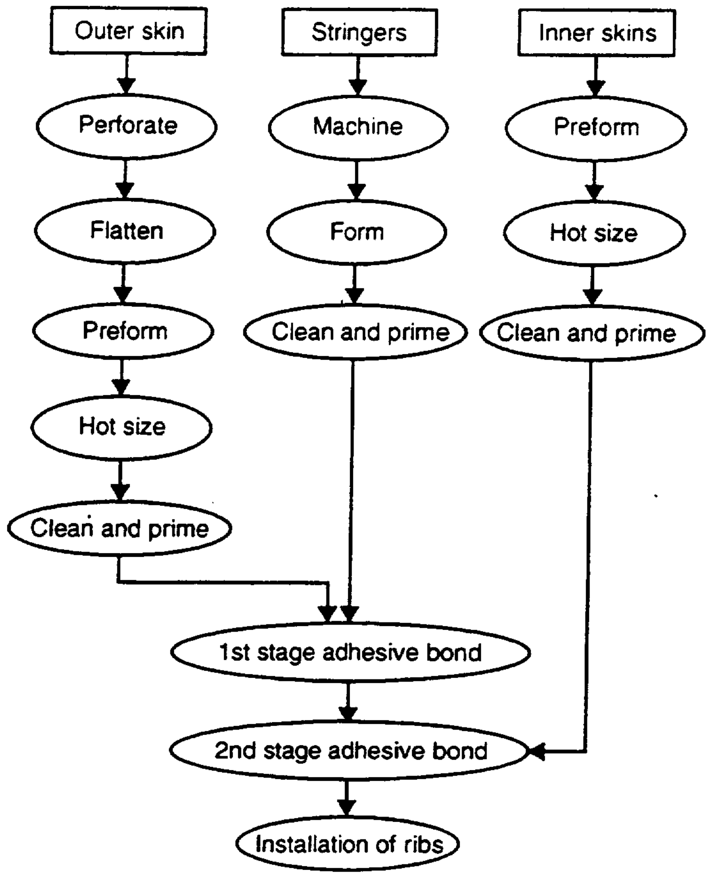

Figure 4.4-5. Suction Panel Assembly Sequence

by foreign object impacts. The skin for the LEFT was welded together from relatively small pieces and had to be flattened before being formed to contour.

The development of laser drilling techniques made it possible to use 0.040-in-thick titanium, which met the Boeing requirement for damage resistance. To exploit this advance, a "Perforated Titanium Scale-Up Plan" was developed. This was both a means of communicating HLFC program needs to potential suppliers and a method of evaluating the quality of the perforated sheets they could provide. Three sets of material were obtained:

a. Lot 1: Small samples or coupons, typically 6 in square, for an initial screening of the vendors. The criteria included the smallest attainable hole size, hole qualities (such as straightness and internal smoothness), variation in hole size and spacing with respect to the nominal values, cleaning required to remove debris left by the drilling process, and other factors related to establishing the perforator's capability to provide consistent quality in material having the thickness and hole size desired for the final lot. In addition, the flow characteristics of suitable samples were evaluated in a flow calibration facility.

b. Lot 2: Larger samples to demonstrate that the vendor could scale up the process developed on coupons to the 3- by 22-ft panel required for the airplane suction panel.

c. Lot 3: The actual panels to be used for the experiment. Two shipsets were procured, one to serve as a backup.

Evaluation of Small Samples.

a. TKS (Aircraft Deicing) Inc., England.

Design of the TKS laser equipment permitted clamping sheet material on a vertical axis, with a fixed laser drilling horizontally. The "x" axis traveled 10 ft horizontally and the "y" axis moved 4 ft vertically. There were options available whereby the laser could be modified to increase the vertical width or a third axis could be adopted. The machine was capable of perforating long pieces by coiling at both ends. Holes of 0.0025-in diameter or smaller, with variable spacing, could be drilled.

Eleven 6- by 6-in coupons were prepared for the initial evaluation, and two coupons were added later. Table 4.4-1 lists the pertinent information concerning the physical characteristics of these coupons.

b. GKN Sheepbridge, England

The GKN laser equipment was a horizontal X-Y table with a laser mounted above the work piece in a fixed position. The laser could be programmed for variable hole spacing on a 3- by 6-ft sheet.

The specification for the scaleup of the perforated titanium was very similar to that prepared for TKS, because it was decided that GKN could perforate the titanium in one piece in a similar manner. Ultimately GKN declined to attempt a one-piece skin and also declined to work toward the development of smaller hole sizes.

GKN perforated ten 6- by 6-in coupons for appraisal during the Lot No. 1 phase. The results of the evaluation of these coupons are presented in table 4.4-2. Figure 4.4-6 shows the effect of hole spacing on the warpage of the titanium sheets and illustrates vividly the significant increase in warpage that occurs as the hole spacing is reduced to 0.015 and 0.010 in (center-to-center). The particular coupons evaluated had a constant row spacing of 0.040 in. The warpage would increase if both hole spacing and row spacing were held equal, as desired for the leading-edge skins.

Suppliers using electron beam (EB) equipment were also evaluated. The results of the entire Lot No. 1 investigation are summarized in table 4.4-3. The limitations of the EB method, noted above, applied to all suppliers. The two laser perforators, TKS and GKN, had the advantages of being able to perforate holes in material of up to 0.040-in thickness and of being able to handle large enough sheets to make welding unnecessary. Therefore, we decided to work with the two laser perforators in proceeding to Lot No. 2.

Table 4.4-1. TKS Perforated Titanium Results

Specimen ID	Thickness, (+), in	Nominal hole diameter, in	Actual hole diameter, in	Coefficient of variation, %	Percent blocked, %	Spacing Rows, in	Spacing Holes, in
1.4	0.036	0.0025	0.00270[3]	20.1	0.2[1] 1.2[2]	0.0096	0.0099
2.3	0.036	0.0025	0.00272[4]	20.1	0.9[1] 1.8[2]	0.0149	0.0149
3.3	0.036	0.0025	0.00287[5]	11.9	0.3[1] 0.3[2]	0.0149	0.0149
4.3	0.036	0.0025	0.00318	9.8	0.7[1]	0.0248	0.0260
5.3	0.036	0.0025	0.00304	15.1	1.2[2]	0.0293	0.0308
5.4	0.036	0.0025	0.00287	14.4	0.7 1.7[1]	0.0294	0.0307
6.4	0.036	0.0030	0.00347	12.6	1.2[2]	0.0246	0.0258
7.4	0.036	0.0028	0.00320	13.3	2.9[2]	0.0245	0.0259
8.4	0.036	0.0024	0.00287	15.3	3.9[2]	0.0247	0.0260
10.4	0.028	0.0025	0.00304	8.9	0.8[2]	0.0245	0.0259
11.4	0.028	0.0025	0.00303	11.9	2.3[2]	0.0245	0.0259
9.4	0.036	0.0021	0.00255	12.6	3.1[2]	0.0243	0.0262
12.1	0.036	0.0010	0.00114	21.9	39.8[2]	0.0099	0.0099

Notes:
[1] Sanded on one side when percent blockage measured.
[2] Sanded on both sides when percent blockage measured.
[3] Entrance side diameter = 0.00408 in.
[4] Entrance side diameter = 0.00424 in.
[5] Entrance side diameter = 0.00409 in.

N30677-023

Table 4.4-2. GKN Perforated Titanium Results

Specimen ID	Thickness, in	Nominal hole diameter, in	Actual hole diameter, in	Coefficient of variation, %	Percent blocked, %	Spacing Rows, in	Spacing Holes, in
3-0.015P	0.033	0.0025	0.00311	15.6	0.8	0.0156	0.0156
3-0.20P	0.033	0.0025	0.00329	11.8	0.1	0.0198	0.0218
3-0.025P	0.033	0.0025	0.00300	8.5	0.0	0.0210	0.0260
3-0.030P	0.033	0.0025	0.00345	8.7	0.6	0.0303	0.0319
4-0.015	0.033	0.0025	0.00309	14.4	1.5	0.0141* 0.0169	0.0158
4-0.020	0.033	0.0025	0.00341	8.9	0.0	0.0217	0.0207
4-0.025	0.033	0.0025	0.00349	7.8	0.2	0.0263	0.0248
4-0.030	0.033	0.0025	0.00342	9.4	0.4	0.0302	0.0307
3-0.010P	0.033	0.0025	0.00299	16.0	0.8	0.0294	0.0307
4-0.010P	0.033	0.0025	0.00316	14.7	0.8	0.0096* 0.0117	0.0307

* Row spacings alternated (every other row).

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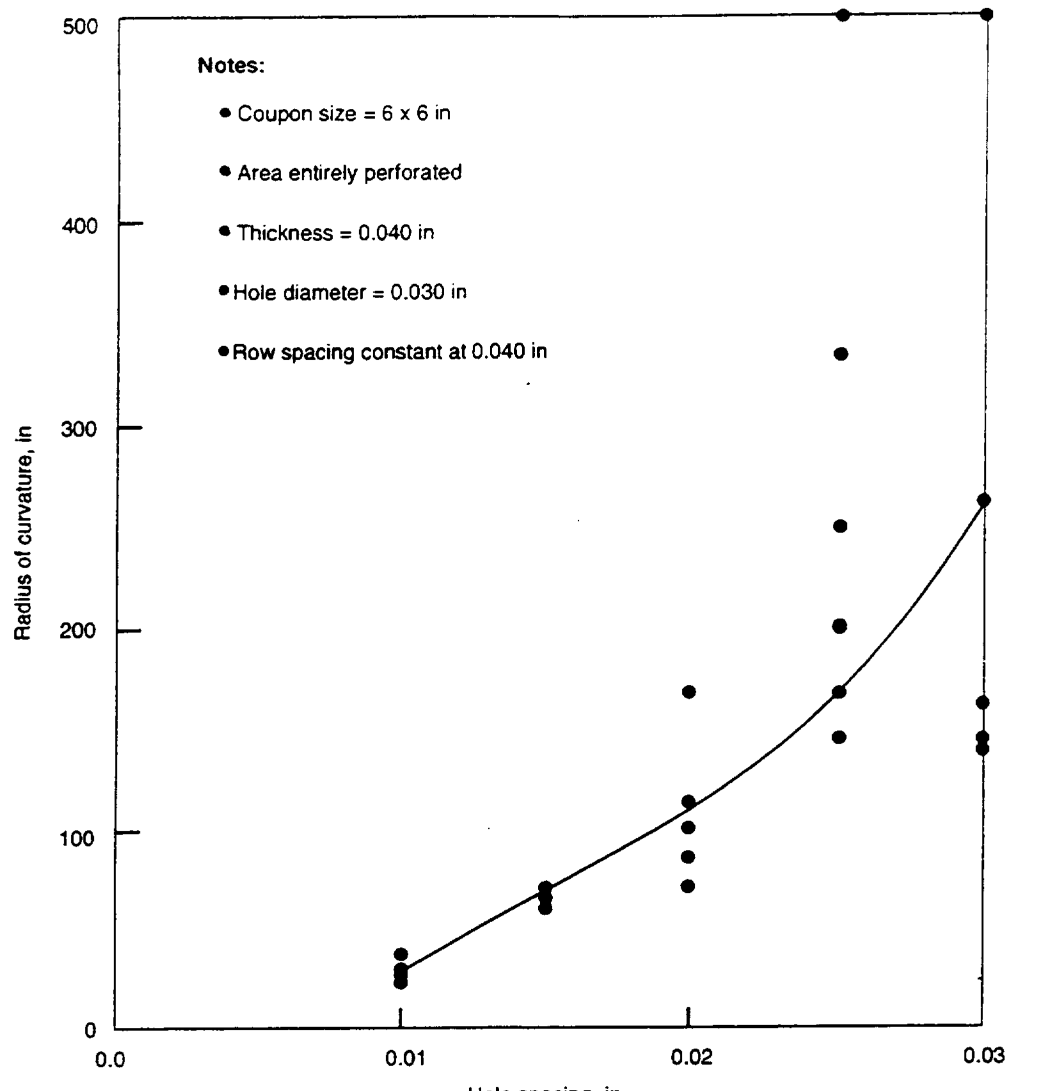


Figure 4.4-6. Warpage Due to Perforating

Table 4.4-3. Evaluation of Suppliers, Lot No. 1

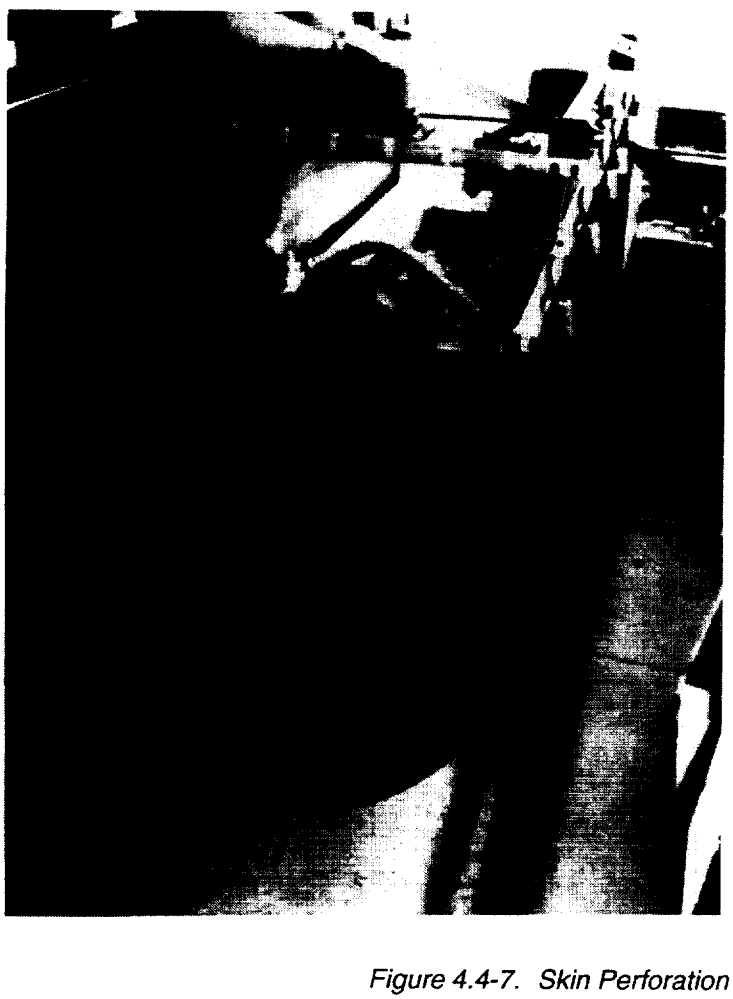
Supplier	EB or laser	Understanding of requirements	No. of sheets	Diameter	Hole quality Blocked	Hole quality Taper
Probeam	EB	High	17	0.0025-0.0030 in	Med	OK
Pratt & Whitney	EB	High	16	Unknown	Unk.	Unk.
Melco	EB	Medium	7	0.0025 in	Med-high	Constrictions
SNECMA	EB	Low	13	0.0035-0.004(Ti) 0.0024-0.0031 (steel)	Unk.	Unk.
TKS	Laser	High	1	0.0025	Low	Marg.
GKN	Laser	Med-high	1	0.0025	Low	Marg.

N30677-025a

Preparation of Larger Samples. Because GKN declined to participate in the large-sample evaluation, the discussion below applies only to the TKS product.

It was initially intended that Lot No. 2 should consist of one large sheet made on the same equipment and using the same procedures and processes as would be used to produce the final lot. It was necessary, however, to perforate four sheets to resolve the difficulties and issues that arose during this part of the development.

Each sample sheet was divided into a number of bays bounded by the rib locations. Flutes within each section were identified by a numerical reference and subdivided for the purposes of airflow/ pressure drop measurement. The first sheet of Lot No. 2 was drilled on a section-by-section basis beginning at the outboard end of the sheet and working inboard one bay at a time, perforating from the top of the sheet down. The laser parameters were set to produce a relatively large but consistent hole diameter. These settings were held constant over the first five bays to give the laser time to warm up and stabilize. Beginning with the sixth bay, the laser parameters were varied to explore the potential for reducing the hole diameter. The full-size sheet of titanium mounted for perforating is shown in figure 4.4-7.



- Laser perforation
- Hole diameter = 0.0016 and 0.0023 in.
- Thickness = 0.040 in. titanium
- Variable spacing

- 30- x 264 in. sheet size

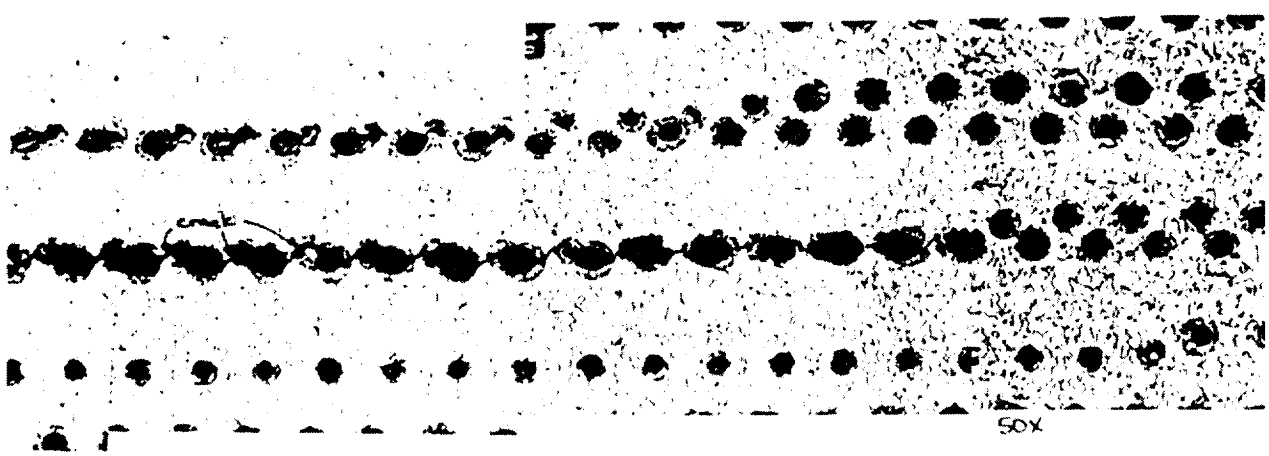
Figure 4.4-7. Skin Perforation Apparatus

N30677-27M

Hole Quality Evaluation. The hole quality was assessed in terms of the numbers of holes blocked by masking a rectangle on the surface to expose 100 holes and visually counting the blocked holes.

Several procedures were evaluated during the production of the full-size sheet. Initially, 10% of the rows were omitted, with the intention of providing rework capability by adding rows if the flow rates were low or uneven. In practice, this procedure would have been very slow, because the laser equipment had to be shut down while suction measurements were made, and, if rows were added, additional suction tests had to be made to verify the final flow rates. It was later decided to abandon this procedure because the control of the hole diameter improved enough that it was no longer necessary to provide such a margin for error.

The first sheet of Lot No. 2 was cut up for experiments on the forming of perforated titanium. A part of the forming process required that a large radius bend be put into the titanium near where the material would drape over the male tool. In the process of forming this large radius bend, it was found that the titanium would crack along the rows of holes in the densely perforated leading edge. It was found that rows of holes were overlapping as a result of rocking of the beam supporting the material during the perforation process. This effect is illustrated in figure 4.4-8.

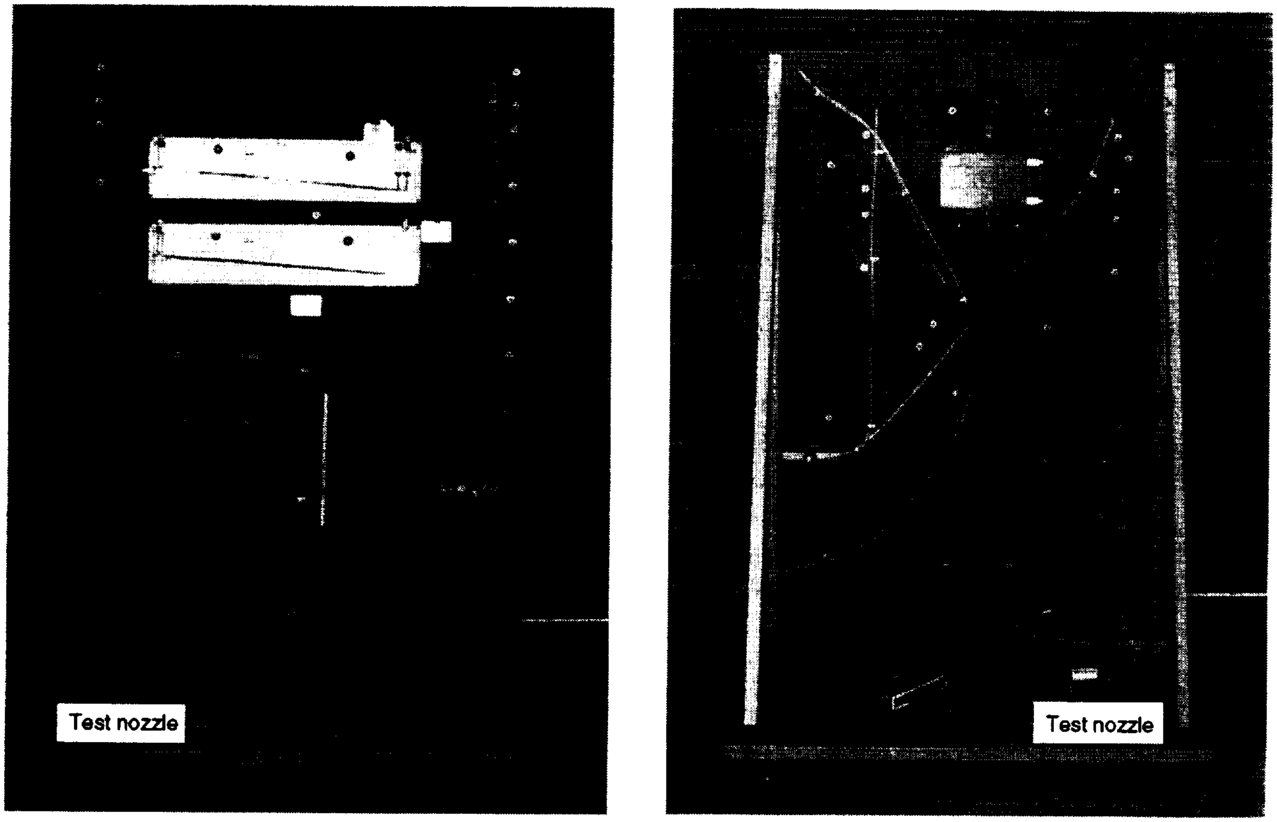


N30677-28M

Figure 4.4-8. Overdrilled Rows

The process of reducing the size of the perforated holes by slowly varying the primary laser parameters resulted in two bays having desirable hole diameters. A range of laser parameters, that could be reasonably maintained for consistent performance, was determined. Bay E (fifth from the inboard end) produced a hole slightly larger than 0.0021-in diameter, and Bay D (fourth from the inboard end) produced a 0.0016-in-diameter hole.

Airflow per square inch was measured at pressure drops of 10 lb/ft^2 and 100 lb/ft^2. The measurements were made using the test rig shown in figure 4.4-9. The measurements on flutes 1 through 14 were made using a nozzle measuring 10 in spanwise and 1 in chordwise. A smaller nozzle, 4 in long by 1 in wide, was used for measuring flutes 15 and 16.



Front View **Back View**

N30677-29M

Figure 4.4-9. TKS Flow Test Rig for Perforated Skin

Figure 4.4-10 shows airflows for bays D and E in comparison to flow rates for earlier test samples. It was subsequently decided to proceed with designs based on these two diameters, with the smaller hole diameter being used along the leading-edge highlight. The availability of this smaller diameter resulted in significant design simplification because of the greater pressure drop that could be maintained across the surface.

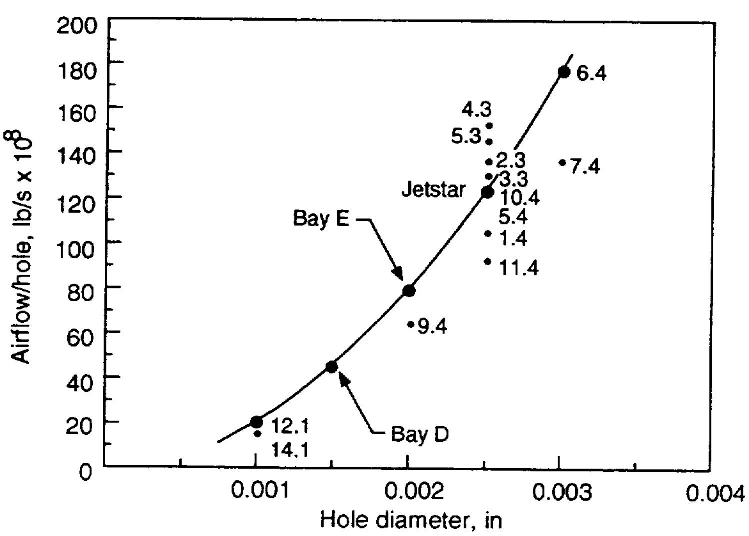


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Figure 4.4-10. Airflow at Choked Conditions

After examining the Lot No. 2 full-size sheet, it was concluded that—

a. The cleaning of the debris from the perforations needed to be improved.

b. The perforations, especially in the highlight region with hole spacing of 0.010 in, were causing warping, making the preforming difficult, and contributing to the over-drilling of rows of holes.

c. The warping and oil-canning of the sheet during drilling caused variations in the alignment of rows of holes and affected the consistency of the hole diameter.

d. The over-drilled rows of holes in the highlight region caused cracking of the sheet during preforming.

e. It was possible to consistently perforate hole diameters of 0.0023 and 0.0016 in.

It was decided that a second phase of the scaleup process was required to address these problems, and that additional sheets should be drilled.

The equipment was modified by the addition of pillow blocks to the carriage guides to reduce the amount of rocking of the support beam. Lot No. 2A was then begun with the following plan:

a. The sheet size was 3 by 5 ft.

b. The target hole diameter was 0.0023 in over the entire sheet.

c. Hole and row spacings were 0.025 in for the main area and 0.010 in for the miniflute area.

d. Flutes 1 through 14 were perforated one section at a time. The miniflute area was perforated horizontally across the sheet in one pass to permit assessment of the possible reduction in warpage.

e. The holes were to be perforated with the laser parameters used in bay E in Lot No. 2, to evaluate the possibility of producing the 0.0023-in-diameter holes consistently.

Cleaning the perforated titanium had two aspects: cleaning debris off the surface and cleaning the crater rim of ejecta around each hole. During the perforating, tiny particles of titanium tended to adhere to the outside of the sheet after being ejected from the hole. A high-pressure hot water spray was found to dislodge the loose titanium particles quite well, as long as sufficient time and patience were used. The removal of the lumps of ejecta was more difficult. Various coatings and cleaning methods were tried with moderate success. The best results were obtained using a power sander operating at low speed with No. 240 grit sand paper. It was important not to hurry the process because the metal tended to smear across the holes if heat was developed in the sanding process. Graphite coatings such as "spray-lube" showed only marginal improvement, and other proprietary coatings improved on the problem only slightly. No coatings were applied in the cleaning technique used for subsequent sheets. Following delivery, the Lot No. 2A sheet was cut up for process development tests.

A third sheet, Lot No. 2B, was similar to No. 2A with the difference that the miniflute area was perforated in rows running vertically on the machine; that is, rows of holes running chordwise rather than spanwise. This was done to evaluate the possibility of reducing the over-drilling of rows because the length of any one row was reduced and variation in spacing became less critical in terms of bending stress. This proved to be successful in subsequent forming experiments.

A fourth sheet, Lot No. 2C, was perforated using the same plan as for No. 2B, with the exception that all holes were perforated using the parameters for the 0.0016-in-diameter holes. This showed that the laser equipment could be operated consistently in that range of settings and the small hole diameters were feasible. This sheet was formed and used for the 4-ft process verification panel.

After these experiments in perforating and the subsequent building of the verification panel, it was concluded that all manufacturing and process procedures had been demonstrated and fabrication of Lot No. 3 could begin.

Perforation of the Flight Test Panels. Three sheets were perforated for Lot No. 3, following the procedures developed in the Lot No. 2 experiments. The hole pattern selected for the final material is shown in figure 4.2-1. The procedure was established to warm the laser up thoroughly before beginning the perforation process and to perforate continuously until the entire sheet was completed. This minimized variation of the laser parameters during drilling and produced a more consistent hole diameter.

The flow rates were checked before and after cleaning. Generally, they were within the tolerances determined as discussed in volume IV, but there was some variability. Where the variability appeared excessive, attempts were made to improve on the smoothness of the flow-rate variations. This was done by etching out the holes having low flow rates, using nitric-fluoride acid. The etching would remove most of the thin leaf of metal that tended to work over the holes during sanding to remove debris adhering to the surface.

4.4.2 Flattening the Perforated Skin

The laser perforation process resulted in oil-can distortion of the titanium skin, as shown in figure 4.4-11. The distortion was primarily due to the tight perforation pattern required at the highlight, and it could not be overcome by processing changes. This distortion was unacceptable because it prevented the proper alignment of the skin in the press for forming operations.

A tooling plate for the full-size skin was obtained after the flattening process had been verified using a 4-ft section of perforated skin. Before attempting the operation on a distorted skin, a full-size unperforated titanium sheet was processed. The intent of this test was to confirm that thermal expansion of the very long part would not degrade the flatness. Any such degradation would be more obvious on an originally flat sheet. Because this test was successful, flattening of the flight hardware commenced.

N30677-31M

Figure 4.4-11. As-Received Perforated Skin Showing Distortion

The flattening process was adapted from a process that had already been developed for forming the skin contour. The skin was first alkaline-cleaned. It was then sandwiched between a 1.0-in-thick ASTM A-36 steel flat tooling plate and a thin (0.032 in) 304 alloy stainless steel sheet, as illustrated in figure 4.4-12. The tooling plate had been face-ground to provide a smooth surface. A hole in the plate provided access for a thermocouple and application of the vacuum flattening force. The stainless sheet was made up from two alkaline-cleaned, 3- by 10-ft sheets welded end to end. The tooling plate was wiped clean with toluene just before assembly. The stainless sheet was welded around its periphery to the tooling plate to make up the flattening retort, as shown in figure 4.4-13. The weld was checked with a soap solution and against a vacuum. A corner of the retort was then reopened on the opposite end from the thermocouple, and argon gas was introduced to displace the air inside.

The corner was rewelded and the flattening retort was then placed in a large oil-fired furnace. Three additional thermocouples were placed on the retort, as shown in figure 4.4-14. A 29-in Hg vacuum was applied to the retort and all connections were checked for leaks. The furnace cycle began with a rampup to between 1000° and 1050°F in 8 hr. This temperature was held for 1 hr. At this point, the vacuum was released, the retort was backfilled with argon to a slight positive pressure, and the furnace was shut down and allowed to cool. After cooling, the retort was opened by peeling back the stainless steel cover sheet, as shown in figure 4.4-15. The flattened skin was checked for waviness. Three perforated skins were successfully flattened using this method.

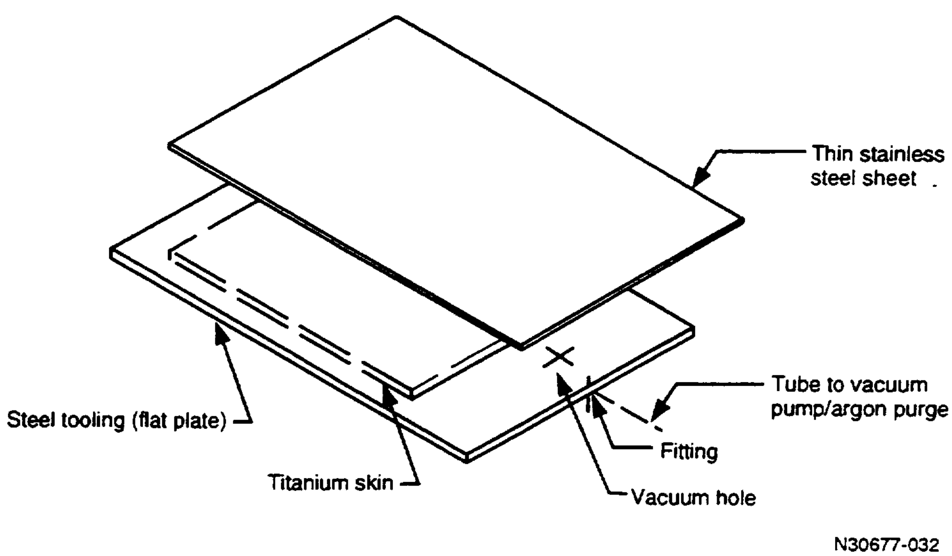

Figure 4.4-12. Flattening Retort Elements

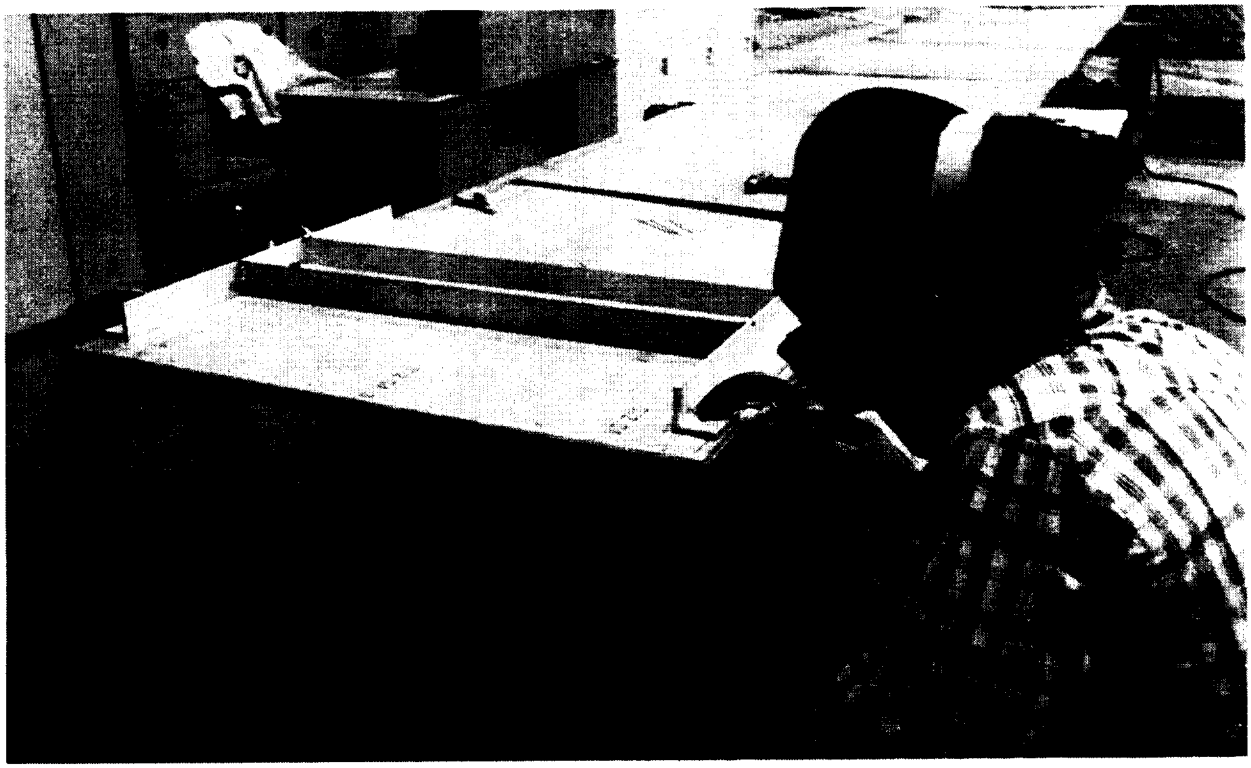

Figure 4.4-13. Welding Thin Sheet to Flat Plate To Make Flattening Retort

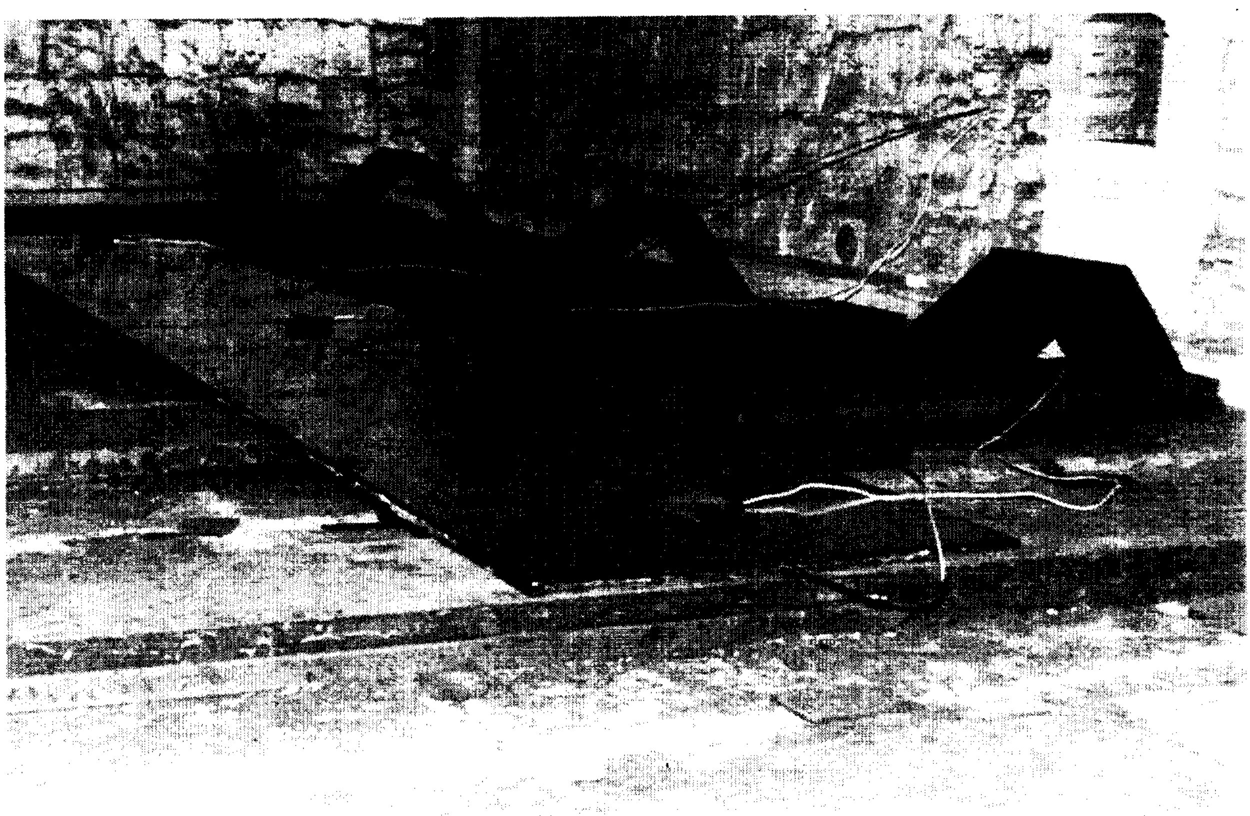

Figure 4.4-14. Flattening Retort Containing Perforated Skin Ready for Furnace Cycle

Figure 4.4-15. Peeling Stainless Sheet Off, After Furnace Cycle, To Reveal Flattened Skin

4.4.3 Preforming the Outer Skin

In order to fit the skin into the hot-forming retort, preforming was initially attempted with a single bend having a radius close to that of the leading edge at the highlight. Problems were experienced in providing forming pressure at the nose, and it was decided to develop the preform contour to conform more closely to the tool surface. The usual methods, chip-forming and roll-forming, were attempted. It was found that the titanium skin had so much springback that the tight radius at the nose could not be successfully roll-formed and that roll-forming left a discrete crease at each stopping point. Chip-forming was tried using a small radius punch and a vee die, which is standard shop practice. This left small creases on the part that, under normal circumstances, would be acceptable. However, the creases violated the HLFC waviness criterion, and it was not expected that hot-forming could remove them. Tests of various combinations of tooling eventually showed that using a large-diameter punch and a trapped uralite-bottom die would produce an acceptable part. A 2.25-in-diameter punch was the largest possible that could form the tightest radius of the leading edge, as shown in figure 4.4-16. The trapped uralite bottom die provided support to the titanium skin to prevent it from leading the punch during forming. Figure 4.4-17 shows these tools installed in a press brake.

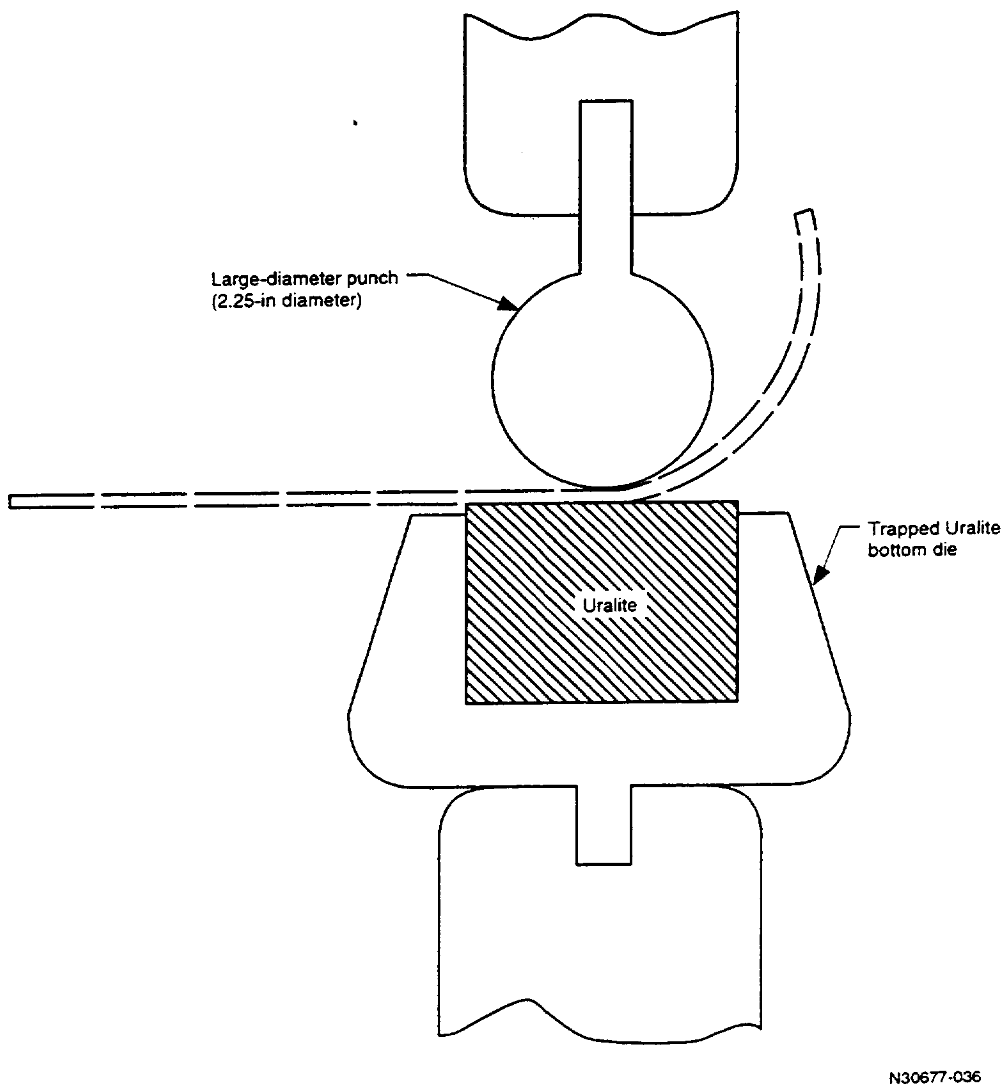

Figure 4.4-16. Skin Preform Using Brake Press

Figure 4.4-17. Preforming Perforated Titanium Outer Skin

The skin was hit at 1/16-in increments for each press setting. The setting was lowered in 0.020-in increments initially, and in 0.010-in increments as the final contour was approached. (Lowering the press setting increased the curvature of the skin at that location.) The preformed skin area was limited to the zone of high-density perforations, about 5 in above and 3 in below the highlight. Because the contour changed along the length, it was checked against specific section templates at each end and at the center of the panel (fig. 4.4-18).

Because of the length of the part and the unpredictable behavior of the skin during forming, this operation was adventurous. Nevertheless, all three skins were successfully preformed. Two were formed initially and the third was held until the results of hot-sizing could be evaluated. It was found that although the first skins were acceptable, a slight crease occurred at a point on the skin where there was a change in perforation density. This crease could not be removed by hot-forming. The third skin was preformed without hitting the area in question. The resulting skin was slightly underformed but still acceptable. This was the skin that was eventually used on the aircraft suction panel.

During preform, generous shimming of the die was required to provide uniform contour. Originally it was thought that a worn press platen was at fault. However, it was found that the shims had to be relocated as part forming progressed and that setups were different for each part. It has since been determined that this was caused by differences among the perforated skins. These differences may have been due to internal stresses, metallurgical or gauge changes within the sheet, or variations in perforation pattern.

Figure 4.4-18. Contour Checking Using Section Templates

4.4.4 Hot Forming Outer Skin to Contour

Preliminary tests were conducted on an existing CH-47 helicopter blade forming tool to establish basic parameters such as forming time, temperature, and vacuum cycles. The tool was modified by the addition of a tee section to the back side of the die. The purpose of the tee was to provide tension to the retort under vacuum in order to apply the necessary forming pressure at the nose. Figure 4.4-19 shows the retort open after one of the four test runs. The original test plan was to try each of three recognized stress relieve/anneal cycles at 800°, 1000°, and 1300°F. This plan was modified after the second test, as contamination became a significant problem. Because 1000°F was found to be adequate for forming, the remaining tests were used to identify sources of contamination. The ceramic fiber blanket that was used to isolate the part from the retort contained a latex binder that would normally burn off harmlessly during heatup. However, trapped inside the retort, this binder was not allowed to escape, resulting in significant contamination and fouling of gauges and valves. A test run without the blanket showed it was not needed.

The final verification tests used a full-size, 4-ft section of the actual flight article size and form. The chosen section had the greatest amount of saddle and also provided a change in profile over its length. At this time a more complex perforated skin was proposed, having variations in hole diameter and spacing. Because of the requirement for positive location of the perforation pattern, additional tool location holes were needed to pin the part to the tool at the highlight during forming. The retort for

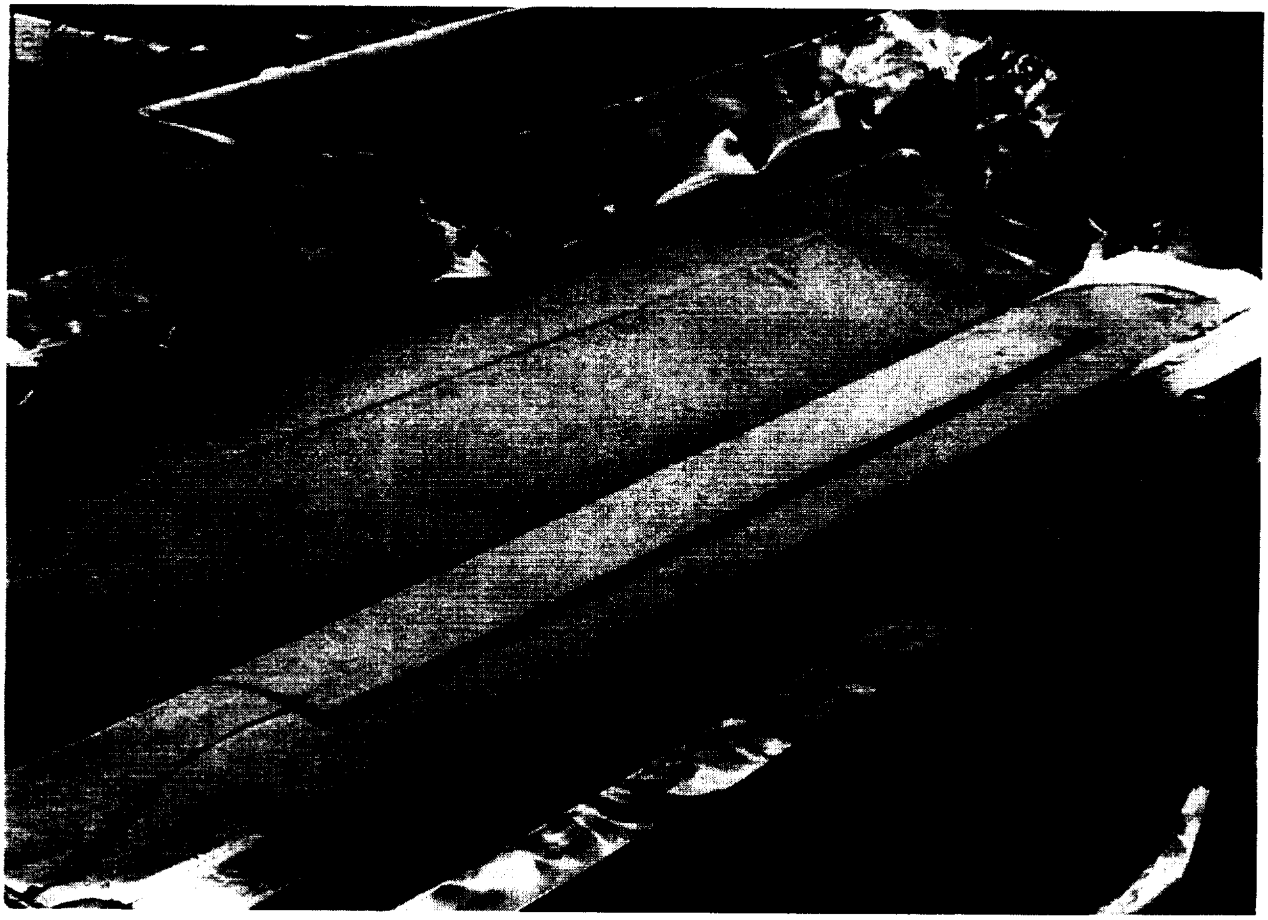

Figure 4.4-19. Preliminary Forming Tests Using Modified CH-47 Form

the verification test was more difficult to assemble than the CH-47 retort because of the thicker section of the tool. It also required an excessive amount of material just to pull the ends together for welding (fig. 4.4-20). During furnace cycling, the retort suffered leakage at the welds and at the tooling pin locations that resulted in further part contamination.

Inspection of the part revealed incomplete forming at the nose. This was due to the large surface area behind the nose contacting the tool first and trapping the part before adequate pressure could be applied at the nose.

The retort was modified with the addition of end caps, as shown in figure 4.4-21, making assembly of the retort much easier and faster. The modified end caps, together with a channel along the back, eliminated the excessive deformation under vacuum and also reduced end loading that might have resulted in wrinkling of the retort and part. In order to provide forming pressure at the nose, a system of clamps was applied to the retort to pull it and the part down to the tool (figs. 4.4-22 and 4.4-23).

The vacuum was then applied to hold the part in position. The first attempt left the clamps on through the furnace cycle, successfully providing force at the nose. However, some of the clamps had given way, causing wrinkles in the retort and the part. In subsequent processing the clamps were removed after vacuum was applied, thus relying solely on vacuum for holding the skin in position during the furnace cycle. This process was successful in forming the skins.

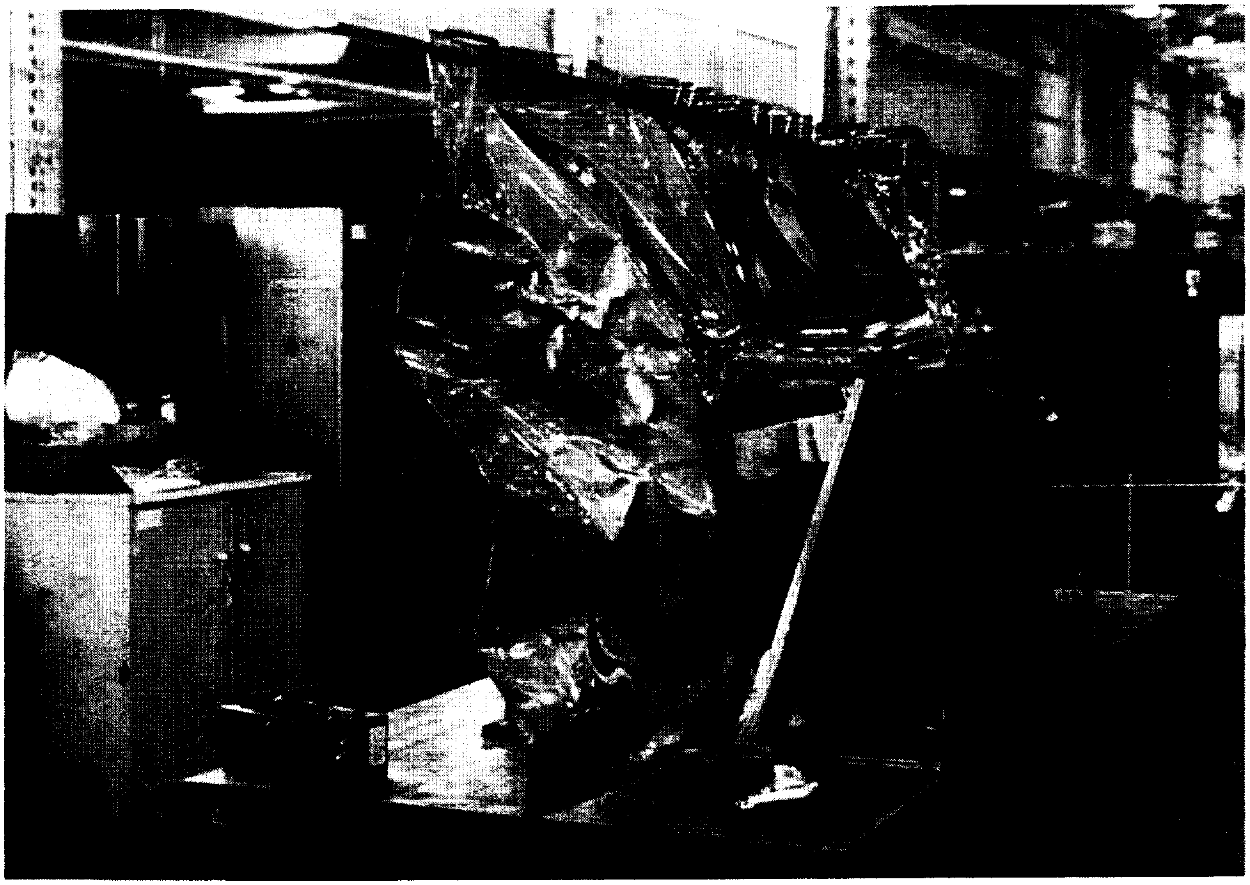

Figure 4.4-20. Retort Used on Initial Forming Tests

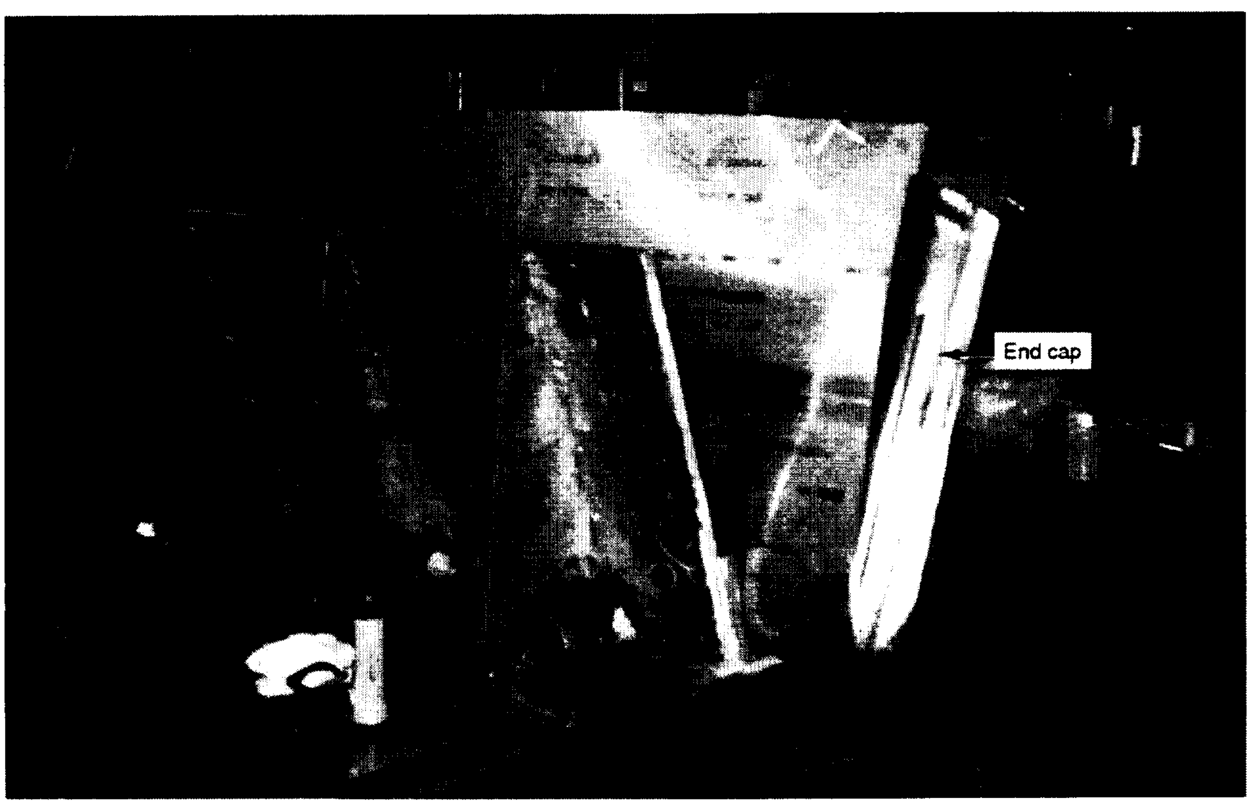

Figure 4.4-21. Modified Retort With Endcaps

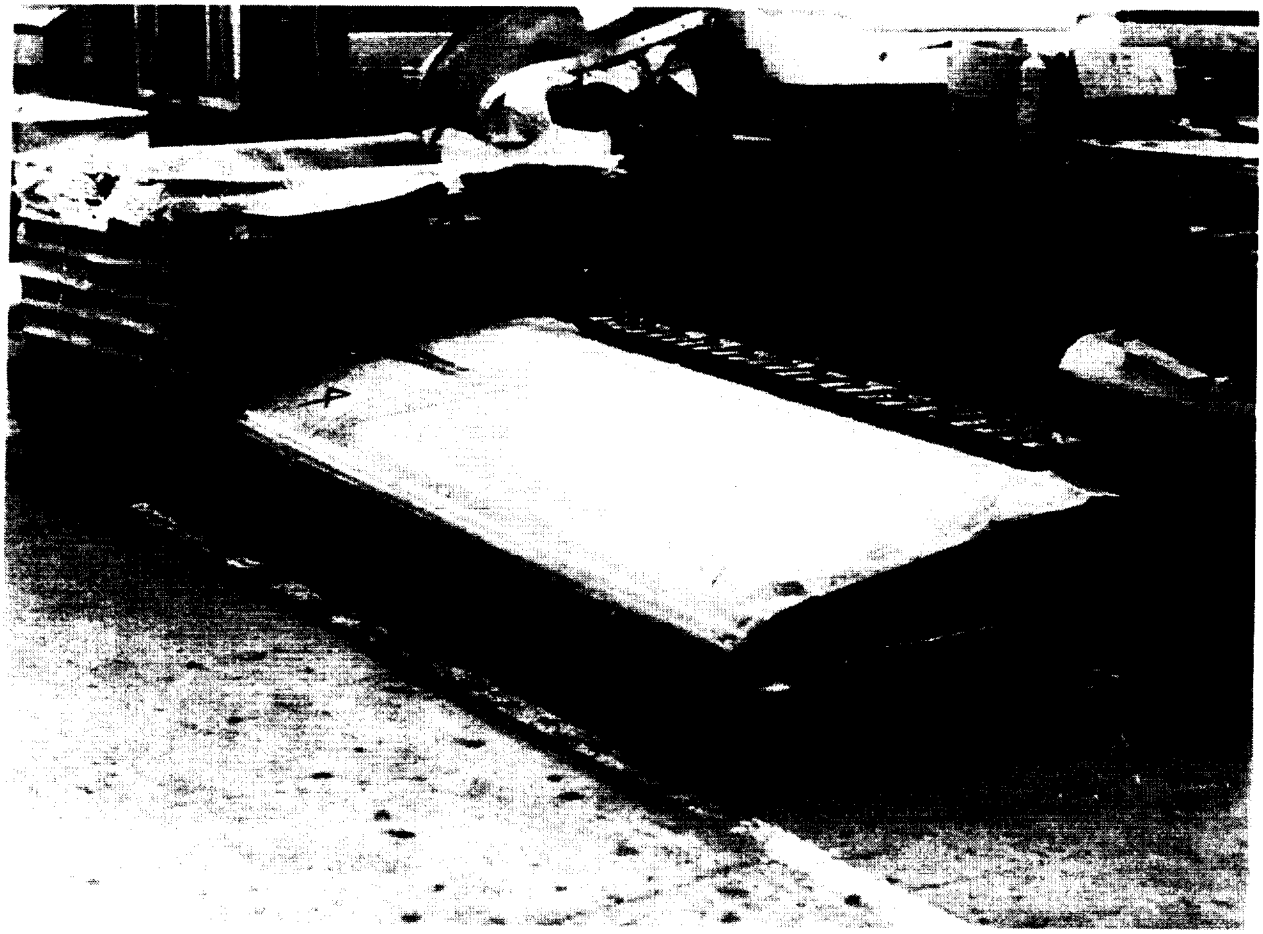

Figure 4.4-22. Modified Retort With Clamps

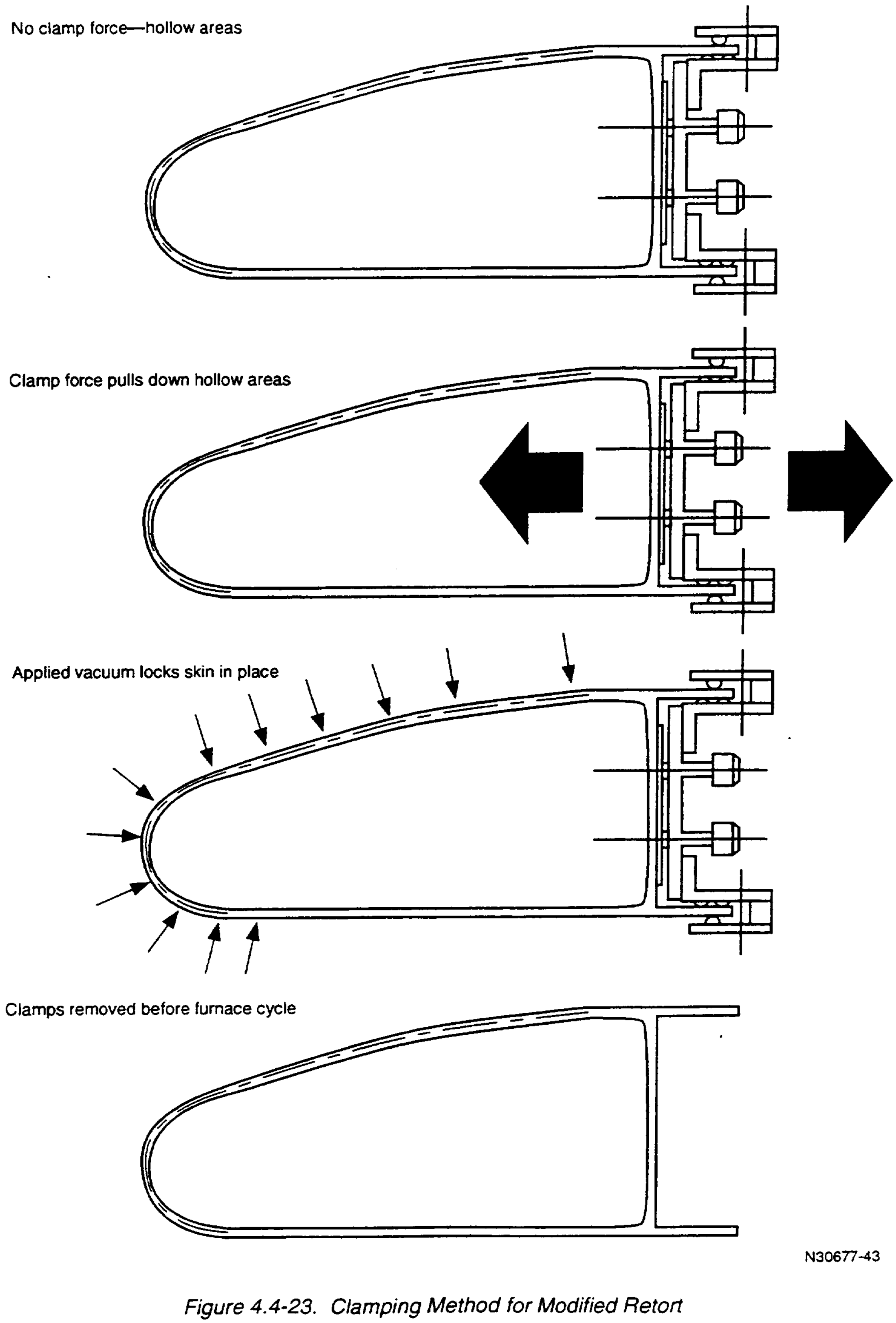

Figure 4.4-23. Clamping Method for Modified Retort

The final selection of tooling for hot forming the perforated skin consisted of a steel form die, forming retort, support cradle, and tensioning clamps. The retort consisted of a large stainless sheet welded up from six 3- by 10-ft sheets, two end caps hydropress-formed to the side profile of the form die, and a brake-formed channel that closed out the back of the retort. All details were 0.032-in 304 alloy stainless steel (fig. 4.4-24).

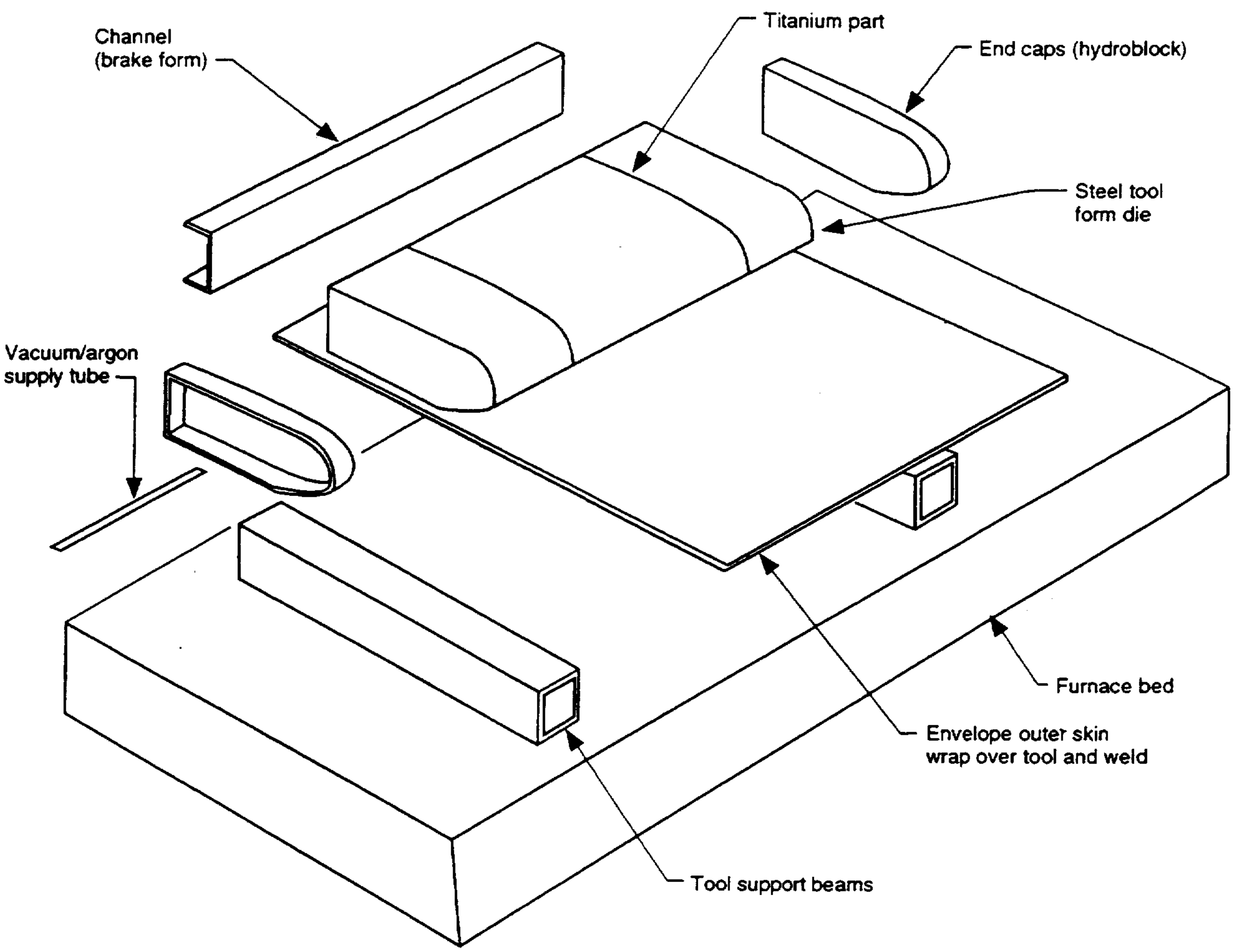

Figure 4.4-24. Skin-Forming Process

The form die was machined from a specially forged 20,000-lb billet of ASTM A-36 steel shown in figure 4.4-25. Machining was programmed from computer data sets generated directly from aerodynamic loft data. The billet was rough-machined and stress-relieved before final machining. Final machining was performed and the tool was hand-finished to meet the waviness requirements. Figure 4.4-26 shows an interim check of waviness during hand-finishing.

After the skin was preformed, 3/8- by 1-1/2-in slotted locator holes were cut into it near each end along the highlight. The skin was then alkaline-cleaned in preparation for hot-forming. The form die and retort details were wiped clean just before assembly of the forming retort. The perforated skin was positioned on the form die with threaded locator pins through the slots at the highlight of

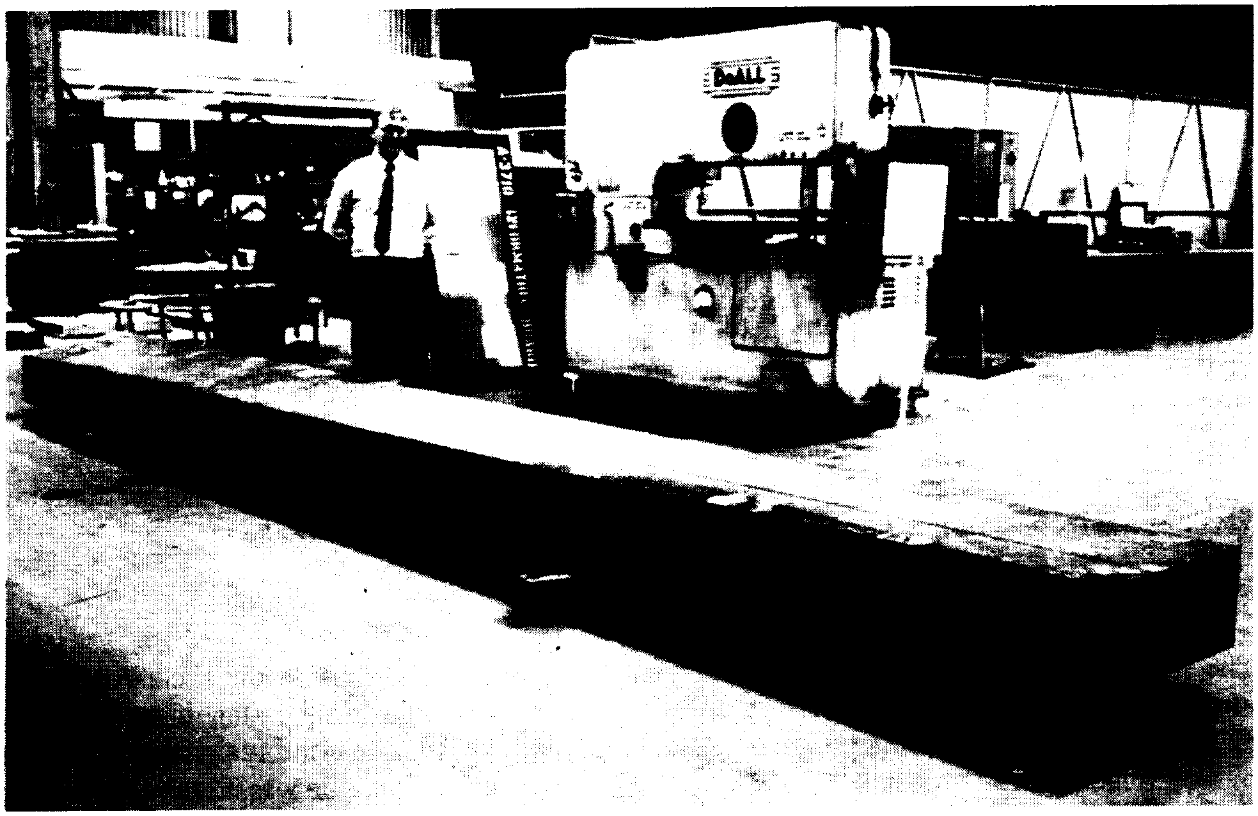

Figure 4.4-25. Specially Forged 20,000-lb Billet for Male Die

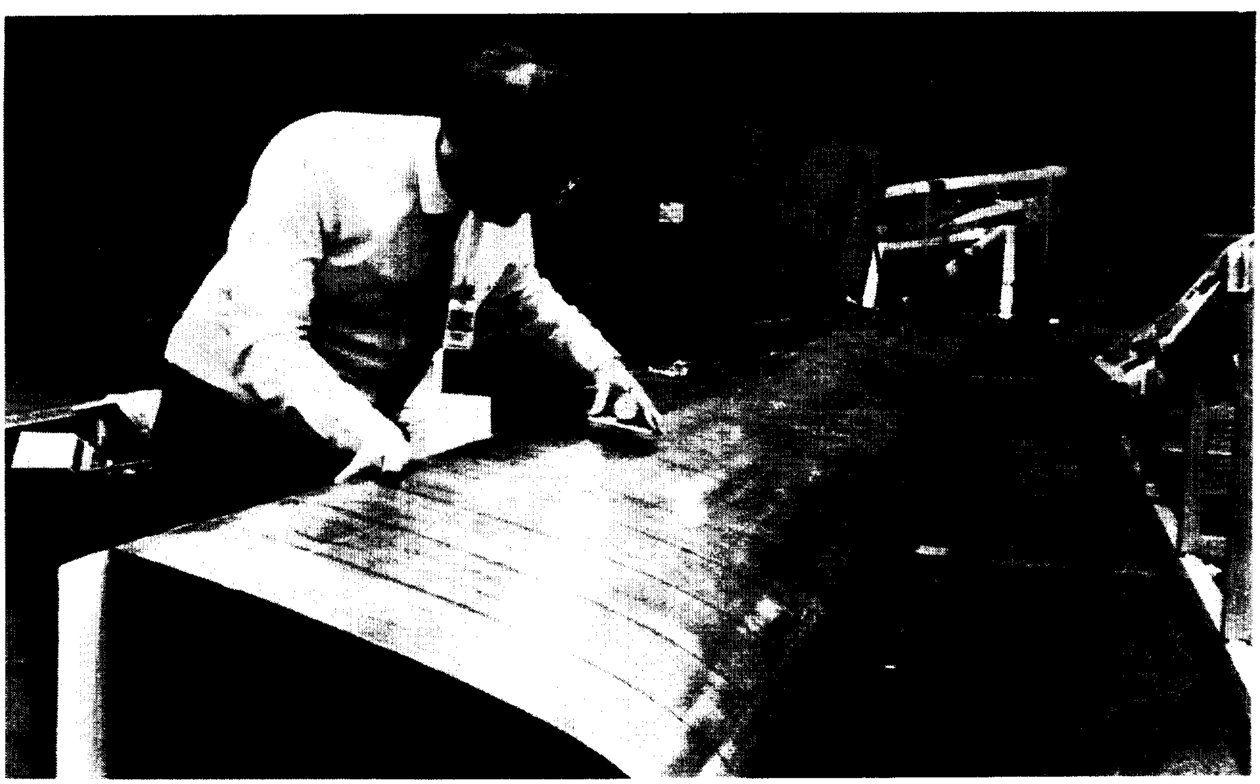

Figure 4.4-26. Measuring Waviness During Handwork on Form

the skin. The form die and skin were then lowered into the retort, already positioned in the support cradle. The end caps and channel for the back of the retort were then fitted and welded in place. The welds of the retort were checked for leaks and the retort was then purged with argon gas. The tensioning clamps were applied to the back of the retort, as shown in figure 4.4-27. Each clamp was tightened until the retort and skin in that area were down hard against the form die. With the clamps in place, the entire retort assembly was transported to the furnace for the hot-forming process. The retort was set up on risers to promote even heat distribution. The forming vacuum (approximately 29 in Hg) was applied to the retort. After a final check for leaks, the tensioning clamps were removed. The furnace cycle was similar to that for the flattening process, ramping-up to between 1000° and 1050°F and soaking at temperature for 1 hr. The vacuum was then released and the retort inflated slightly for the cooldown. After cooldown the retort was opened, as shown in figure 4.4-28.

Contamination was reduced by performing an argon purge of the retort for 8 hr before applying vacuum and by backfilling the retort with argon at the end of the soak period.

4.4.5 Stringer and Inner/Inboard Skin Fabrication

The stringers were machined from commercially pure titanium plate with the exception of the aftmost stringer, which was Ti-6Al-4V alloy. The stringers were formed after machining and checked in a fixture for twist and contour, as shown in figure 4.4-29. Because any significant preload of the stringer was expected to translate into waviness on the outer skin, care was taken to ensure that these stringers were as close to nominal contour as possible.

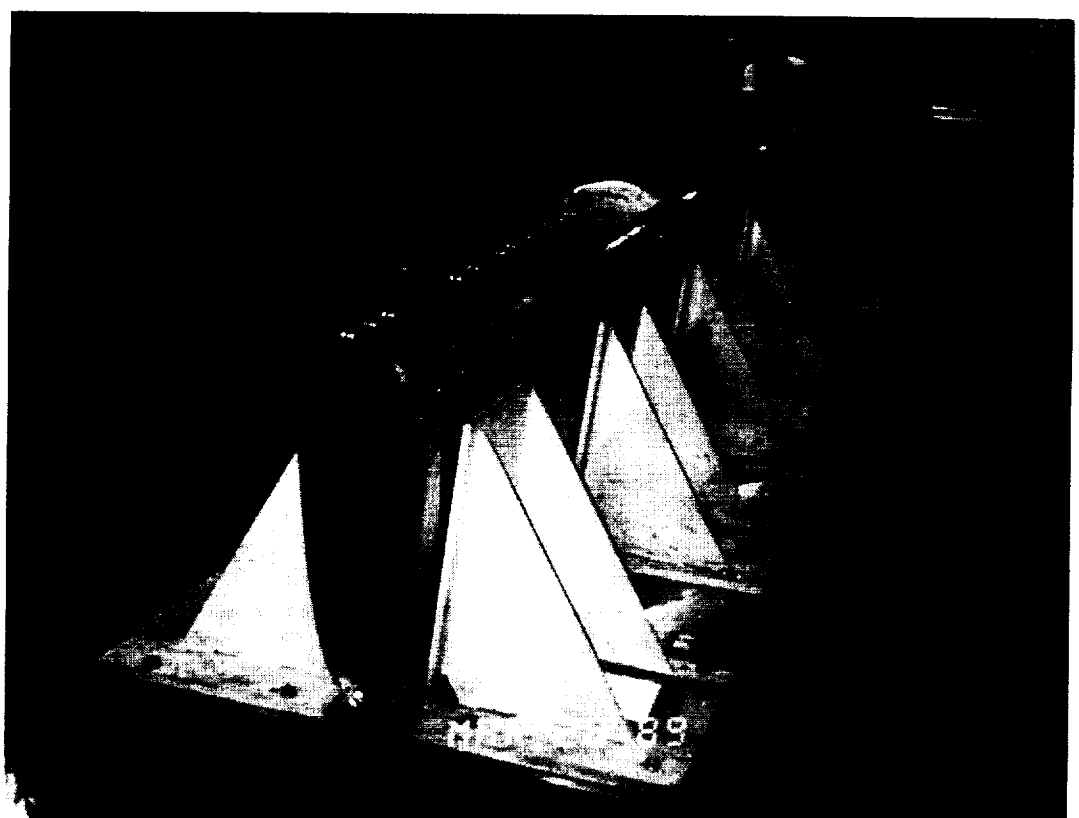

Figure 4.4-27. Form Die and Skin Assembled Into Retort

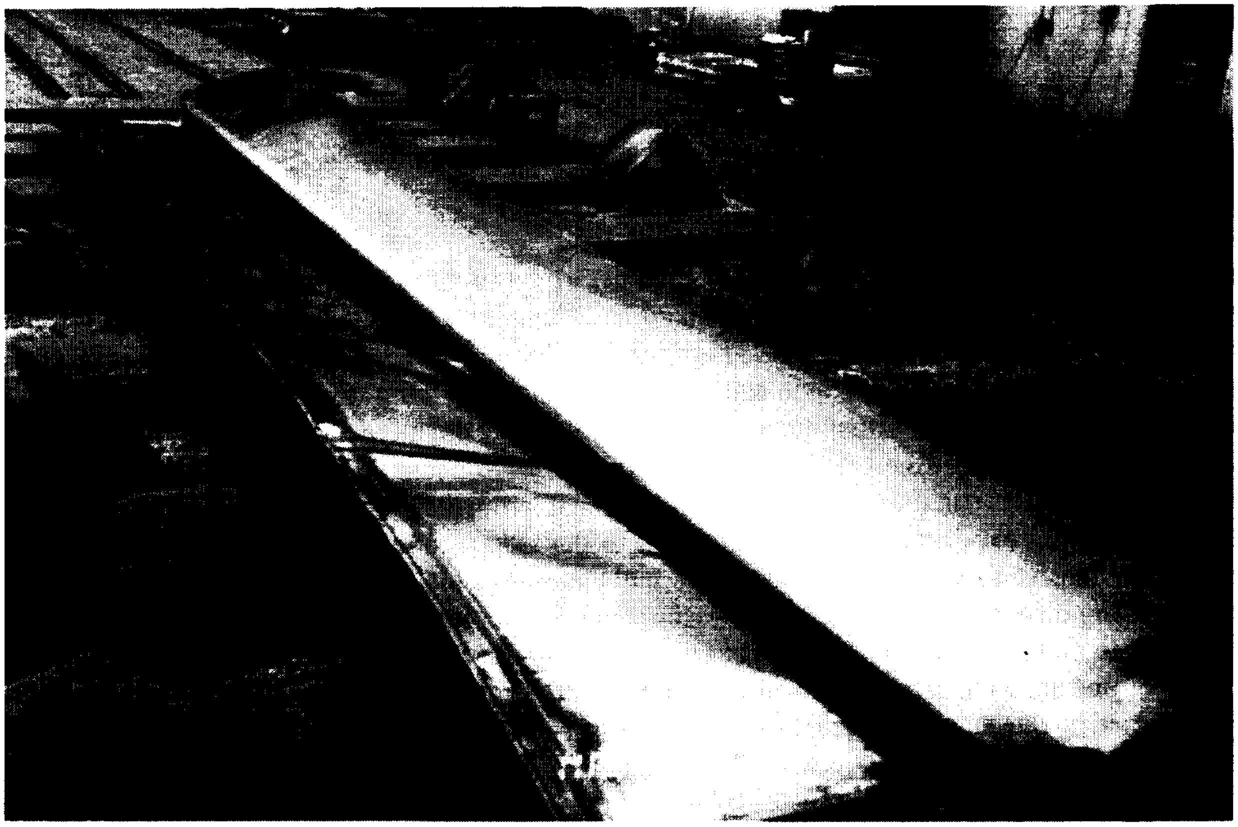

Figure 4.4-28. Retort Peeled Open After Furnace Cycle To Reveal Formed Skin

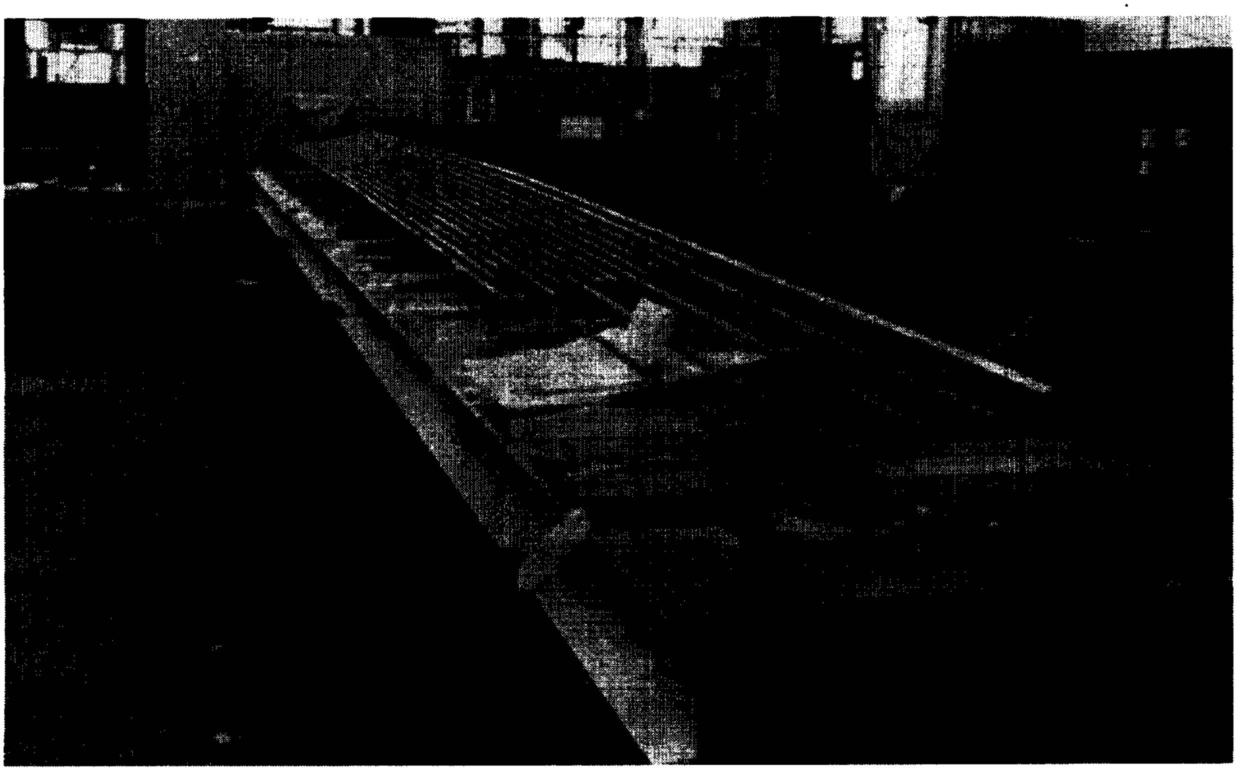

Figure 4.4-29. Fixture for Testing Stringer Twist and Contour

The inner and inboard skins were formed using the same techniques and tooling as the perforated outer skin. One of these skins was formed before forming the outer skin as a final tryout of the process.

4.4.6 Suction Panel Assembly

Adhesive Selection. Adhesive screening was performed in three phases. Phase one consisted of system screening and manufacturing evaluation. Phase two included property evaluation and maximum temperature determination. Phase three included verification of mechanical properties. In addition, process tests were conducted to determine the minimum stringer face width that could be bonded and inspected. Further details of adhesive bonding develoment are presented in appendix A. This work was performed during the preliminary design phase when it was believed that face widths as small as 0.10 in might eventually be required to prevent flow recirculation. A large number of small bonding experiments were carried out to develop techniques for narrow bonds without excessive adhesive flow. These consisted of small panels to which stringers of varying widths were bonded using differing widths and thicknesses of adhesive. It was also considered possible that a third skin to control pressure drop might be required in the first few inches of the suction panel. Details of one such concept are shown in figure 4.4-30. The outer flutes, only 0.10 inch in width, were formed by the channels machined into the pressure drop skin. These were successfully bonded without clogging perforations between the 0.10-in stringers. However, the need for such extreme narrowness in the flutes was eliminated by the development of techniques for perforating the outer

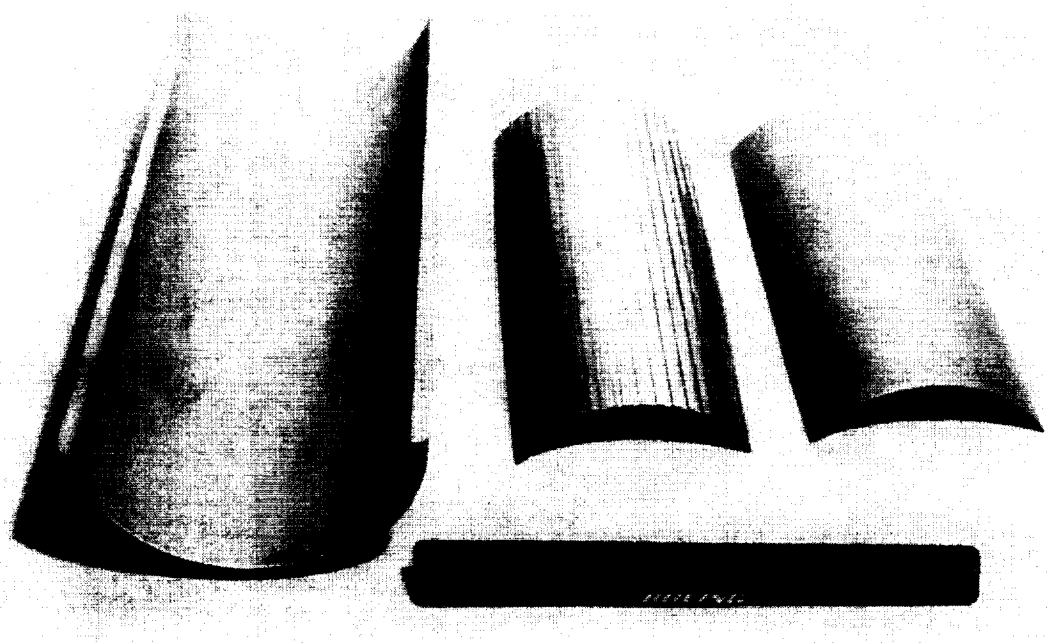

Figure 4.4-30. Experimental Concept for Fabricating Very Narrow Flutes

skin with holes only 0.0016 inch in diameter. As finally manufactured, the minimum flute width on the suction panel was 0.30 in. Bond quality was established by a mechanically scanned, through-transmission ultrasonic inspection method.

In the final design configuration, the skin perforation patterns were different from those previously tested. The porosity and curvature of the skin had a significant effect on squeeze-out of the adhesive during cure. The final stringer configuration and adhesive width determinations were based on the perforation patterns used on the flight article. As a result of these tests, the full-size stringers were chamfered and the adhesive cut back 0.030 inch per edge on the larger stringers.

Skin/Stringer Assembly. The suction panel was bonded in a two-stage process using a modified epoxy adhesive with a 350°F cure. The first stage in the process bonded the stringers to the outer skin. The second stage bonded the inner skin to the stringers. All details were cleaned and primed before bonding. Details were prepared using a phosphate conversion coating process. This standard process included an alkaline-clean, 10-sec immersion in nitric-fluoride acid, and immersion in phosphate-fluoride solution. The perforated outer skin required modification to the process to protect the inside of the holes. Mylar adhesive tape was applied to the outer surface of the skin during the processing. This prevented the acid from reaching the outer portion of the holes where the diameter was most critical. The tape was then removed and the skin was washed using a high-pressure water wash on the outer surface of the skin.

Following the phosphate conversion coating process, all details were primed using B. F. Goodrich PL502 structural adhesive primer with 15% solids. The primer was thinned to 5% solids before application. For the perforated outer skin, a nylon pressure bag was fitted to the exterior of the skin before priming, as shown in figure 4.4-31. While the primer was being sprayed on the inner surface, 1-psi air pressure was applied to the bag on the exterior on the skin. The pressure difference resulted in airflow through the skin, and it kept the holes from being plugged. Pressure drop measurements were made before and after priming. They showed no appreciable change as a result of the processing.

The tools used for bonding the panel included a two-piece steel female bond assembly jig (BAJ) (shown in fig. 4.4-32) and locating fixtures for the stringers. The BAJ was also used to hold the panel for subsequent installation of ribs, instrumentation, and ducting. This construction is considered standard for bonding applications, with the exception of the surface waviness requirements. The finishing of the tool surface was accomplished in the same manner as for the forming tool and to the same waviness requirements. A vacuum check was performed on the joint in the tool to test for leakage.

As originally designed, the BAJ was to be a fiberglass structure laid up from the surface of the forming tool. The intent was to avoid a two-piece bonding tool and eliminate the possibility of hand-finishing the two tools to different final configurations, which might have induced waviness in the panel. The fiberglass BAJ built for development efforts suffered from warpage while being cured. The surface was damaged during subsequent operations, and a skin formed in this fixture reflected the warpage and damage. Finally, the durability of this form of tooling was questionable. Because of these perceived risks with fiberglass tooling, steel construction was chosen for the flight article tools.

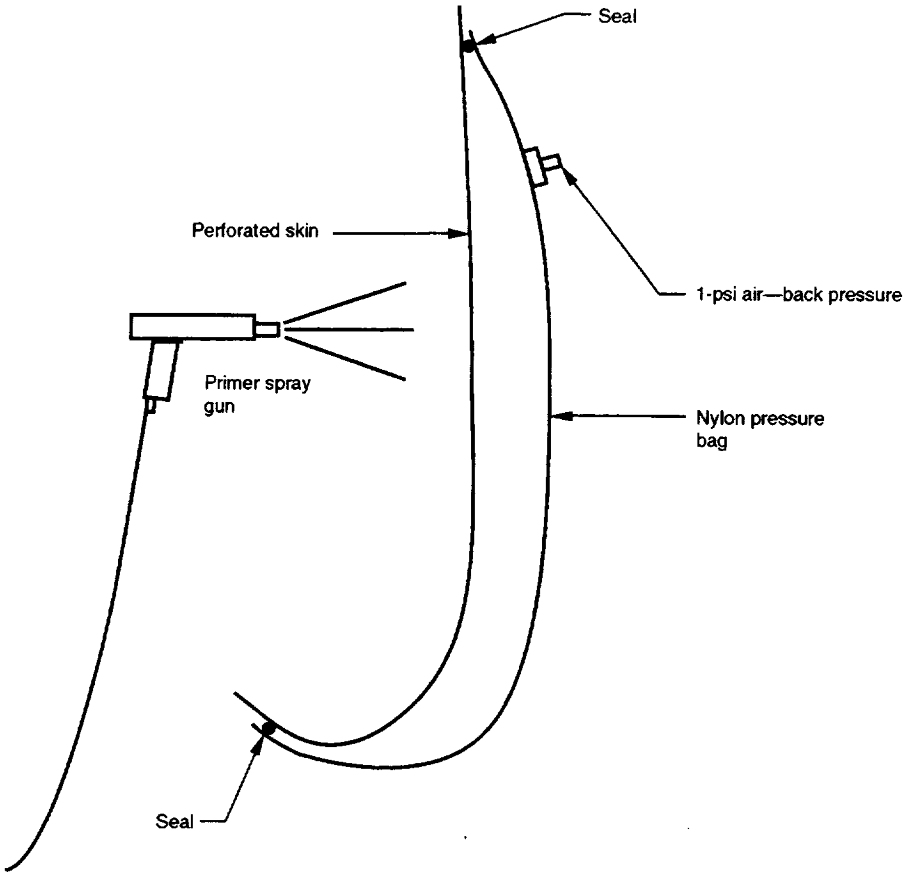

Figure 4.4-31. Primer Application Technique for Perforated Skins

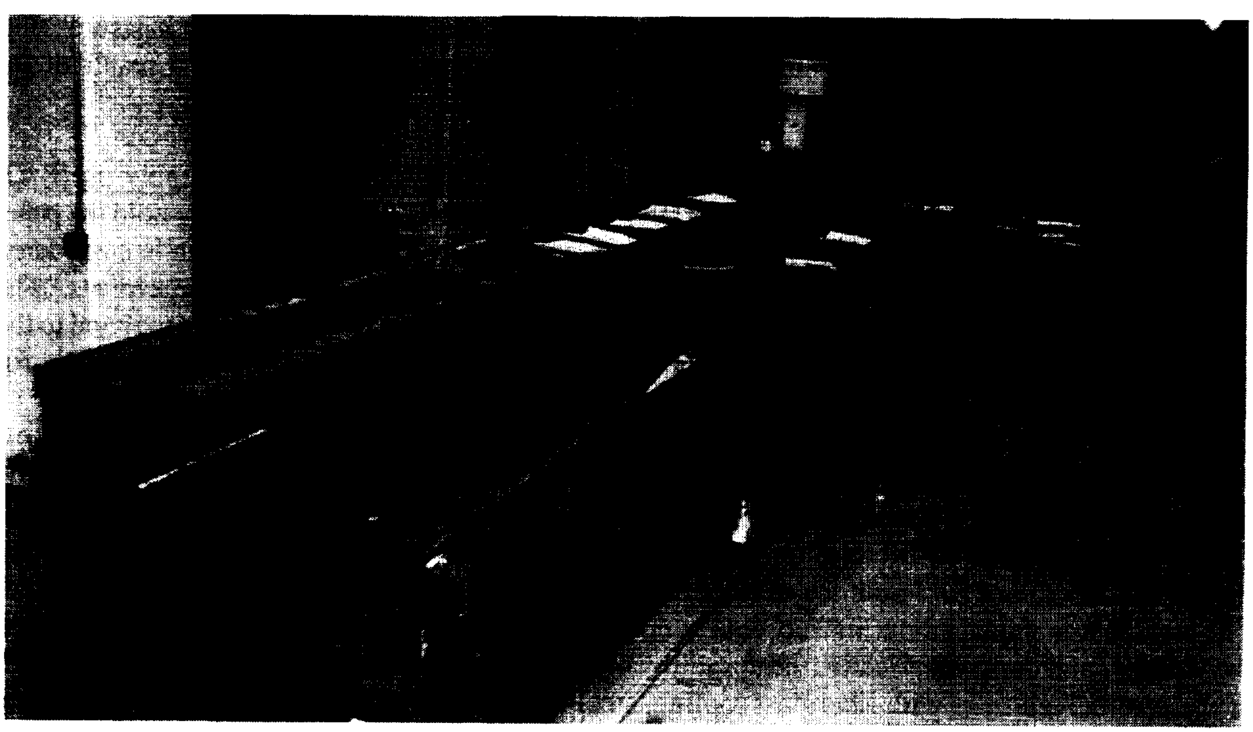

Figure 4.4-32. Two-Piece Steel Bond Assembly Jig Undergoing Dimensional Inspection

The outer skin and stringers were first checked with the BAJ to ensure a proper fit. To ensure that no binding or other unforeseen problems would interfere with bonding, a "dry run" of the bonding process was performed. A layer of adhesive sandwiched between thin mylar sheets ("verifilm") was used. The tool was bagged and autoclave-cured to verify the proper operation of the locating fixtures.

In preparation for the first-stage bond, the outer skin was located in the BAJ with tool pins through slots at each end of the skin. Adhesive was cut into strips of the proper width and heat-tacked to the stringers. The adhesive was trimmed around all instrumentation holes through the stringers. For the full-size stringers (Nos. 1 through 18), adhesive was cut back 0.030 inch from the chamfer on each edge of the stringer. For the smaller stringers, adhesive was cut net to the width of the stringers because testing had shown very little adhesive flow for bonding to the highly perforated skin in the forward section. The stringers were then located on the skin using locating headers as shown in figure 4.4-33. The locating headers had spanwise slotted holes to allow for differential expansion during the cure cycle. Silicone rubber plugs were inserted into each of the instrumentation holes in the stringers, followed by Teflon plungers, as shown in figure 4.4-34. The plungers stood slightly above the stringers. Caul plates were then fitted over the stringers, completing the assembly for the first-stage bond. The skin and details were vacuum-bagged and autoclave-cured. The panel was then visually checked for adhesive voids and to ensure that instrumentation holes were not plugged.

Figure 4.4-33. Stringer Locating Headers

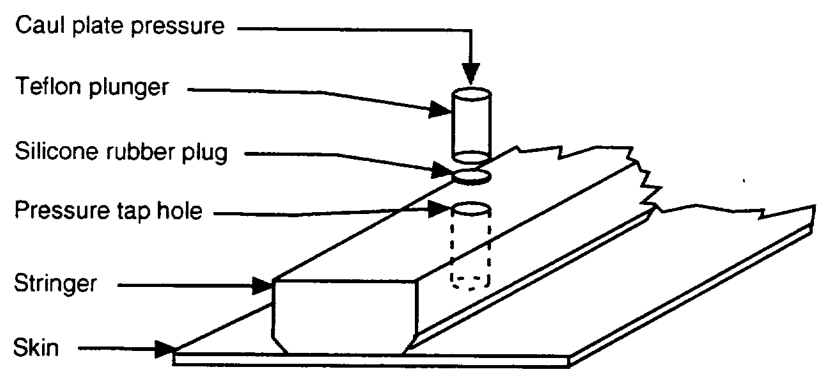

Figure 4.4-34. Silicone Rubber Plug Teflon Plunger Arrangement

Details for the second-stage bond included the forward and aft inner skins, as shown in figure 4.4-35, and parts to seal the panel between the stringer ends and around hinge cutouts. The panel and these details were assembled in the BAJ with a verifilm adhesive sheet. Additional caul plates were positioned over the inner skins to help bridge the unsupported span between stringers. The assembled panel was bagged and autoclave-cured to verify required adhesive thicknesses. The second-stage bond was performed in the same steps as the verifilm cycle using adhesive thicknesses based on verifilm results.

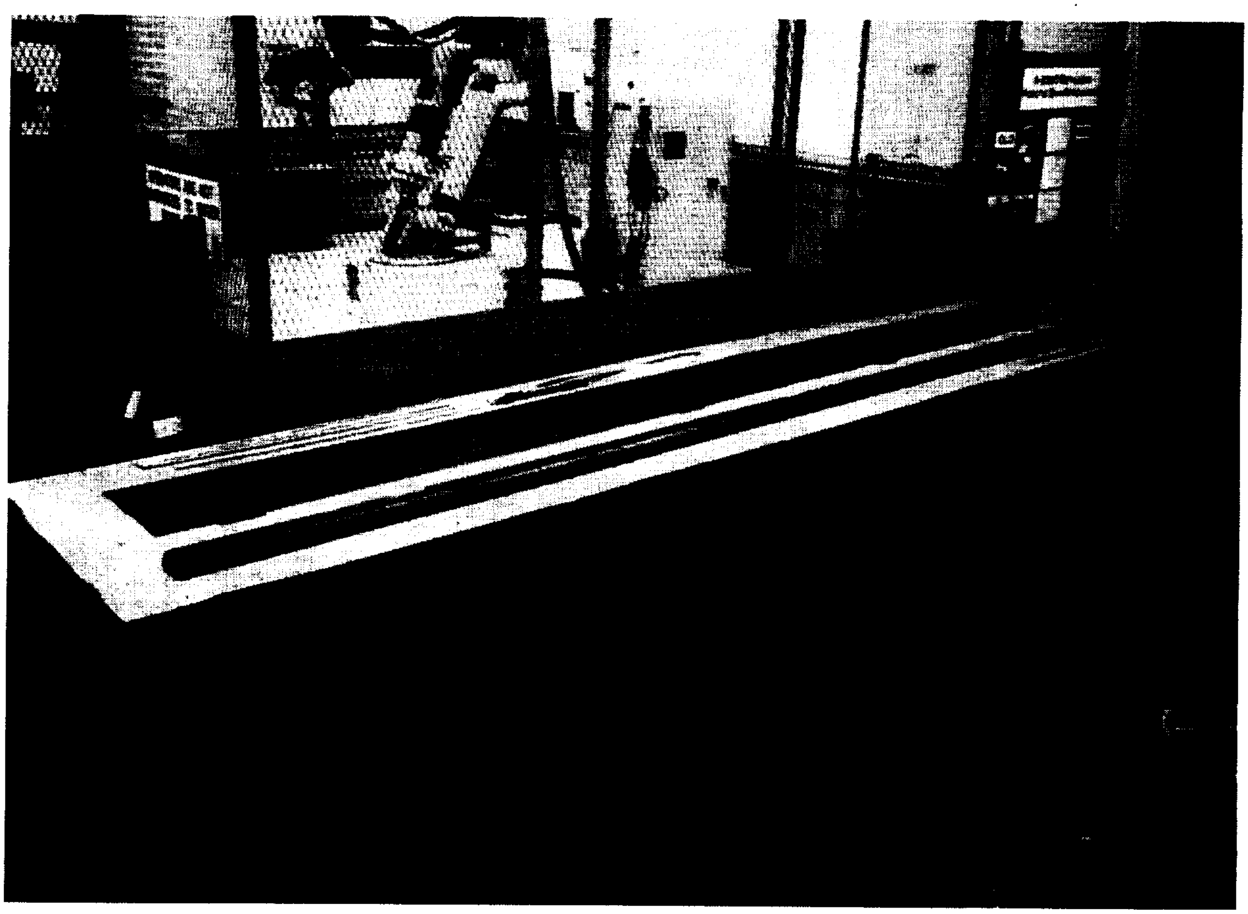

Figure 4.4-35. Second-Stage Bond Details

The panel was checked after bonding and found to be within aerodynamic tolerances of 0.002 inch in 2-inch waviness and within 0.020 inch on overall wing contour.

Installation of Rib Chords to Suction Panel. Installation of the support rib chords to the suction panel was accomplished while the suction panel was still in the BAJ to reduce the likelihood of inducing waviness to the suction panel. All rib locating fixtures were coordinated during construction of the BAJ. A castable epoxy shim was used between the rib chord and suction panel to ensure closest possible fit. The rib installation sequence is shown in figure 4.4-36. The rib locating fixtures

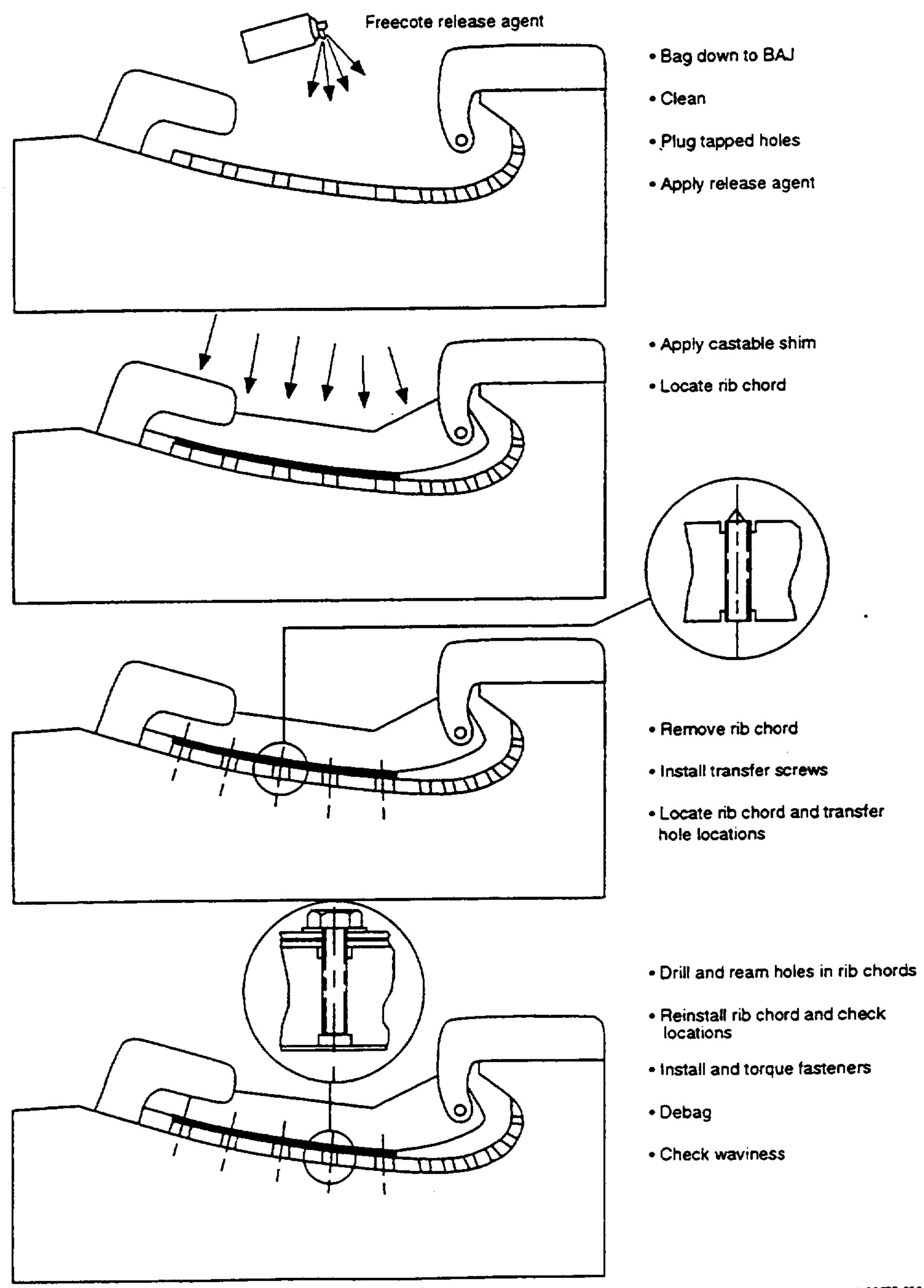

Figure 4.4-36. Rib Installation Sequence

are shown in figure 4.4-37. The suction panel after installation of ribs and suction plenums is shown in figure 4.4-38.

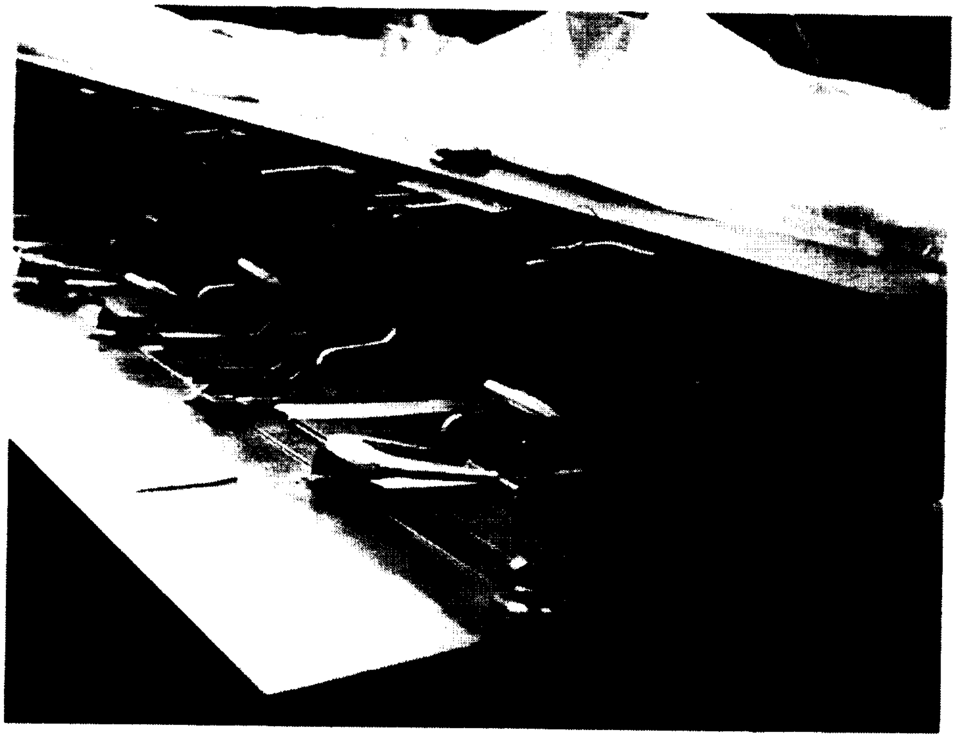

Figure 4.4-37. Rib Locating Fixtures

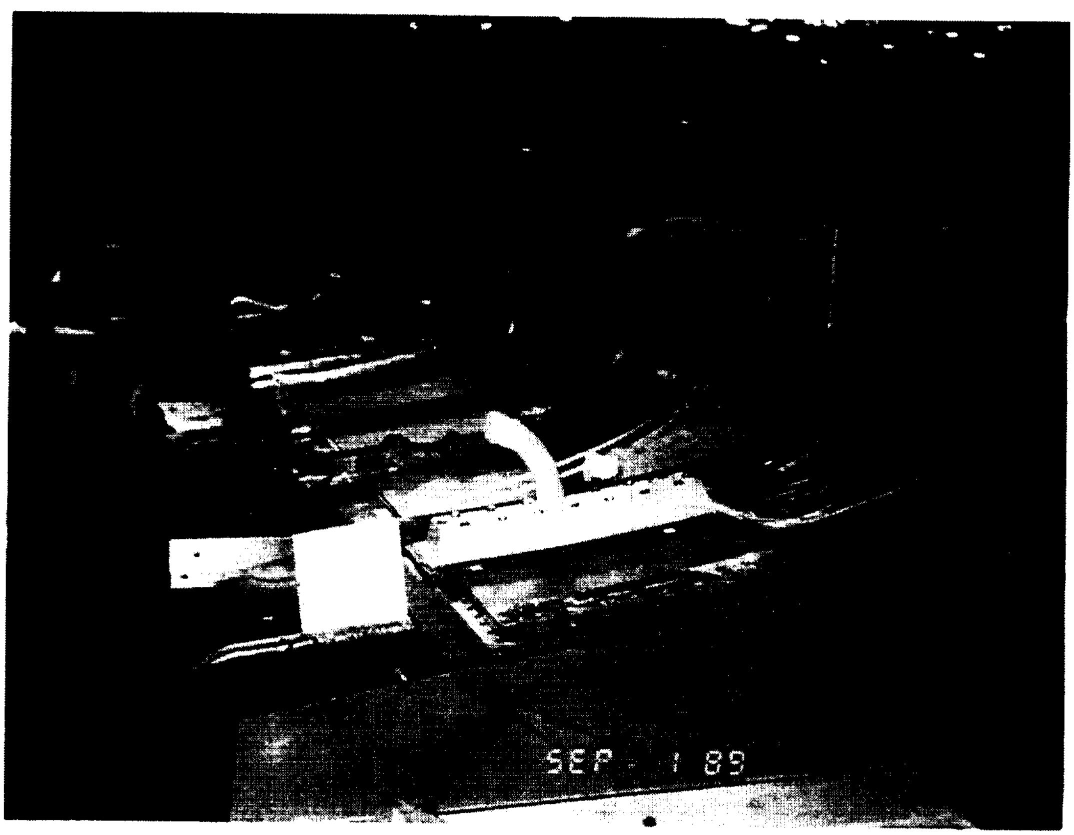

Figure 4.4-38. Suction Panel After Installation of Ribs and Suction Plenums

4.5 PANEL SUPPORT STRUCTURE

Closely spaced supports were needed to meet the requirement for a smooth, wave-free leading edge under varying load conditions. Twenty-one support ribs were provided: one at each end, two at each of the four Krueger flap hinge positions, and eleven intermediate airload ribs interspersed among them. The resulting spacing ranged from 11 to 18 in and, together with the close internal stringer arrangement, it provided the needed surface stiffness.

A typical airload rib is shown in figure 4.5-1. It was composed of a titanium upper chord plus an aluminum plate web and lower chord. The upper chord was an angle section machined from Ti-6Al-4V bar. The aft end had expanded tabs to receive the attachment bolts tying it to the upper wing skin. The web was machined from 7075-T6 plate. A tab was also machined into the lower aft corner to receive the attachment bolts into the lower wing skins. The web and lower flange were combined into a plate web 0.50 in thick, which was then further reduced to receive the fasteners from the rib chord and spar attachment angle. Holes penetrated the web for electrical cables, air ducts, and the flap drive shaft. A clip connected the highlight stringer and the lower aft panel stiffener directly to the rib web. Machined angle members connected the rib webs to the front spar webs.

Each Krueger flap hinge support rib was a three-piece assembly consisting of two webs machined from aluminum plate attached to a channel-section titanium upper chord. The webs were machined from 7075-T6 plate. Double flanges were machined into the lower aft corners to receive the attachment bolts from the wing lower skin. The web was pocketed and machined to receive the bearings for the flap actuator support flange. The channel chord was machined from Ti-6Al-4V and included lugs for the flap hinge bolt at its forward end and flanges for the attachment bolts tying to the upper skins at the aft end. Figure 4.5-2 shows a drawing of a typical hinge support rib assembly. The assembly housed the flap drive shafting and straddled the flap drive linkages.

The end supports provided only vertical restraint to the panel. A shear connection near the nose of each rib allowed panel spanwise expansion or contraction with respect to the surrounding structure. At the inboard end a shear fitting was attached to the existing nacelle rib and slotted to receive a cantilevered rectangular bearing piece. The bearing piece was attached to a partial rib fixed to the suction panel structure. A cross section is shown in figure 4.5-3.

At the outboard end, a special closure rib carried the flight loads from the panel back to the front spar. This rib was slotted and loaded by a rectangular bearing piece attached to a partial rib fixed to the suction panel structure. A view of the rib and a cross section through the connection is shown in figure 4.5-4.

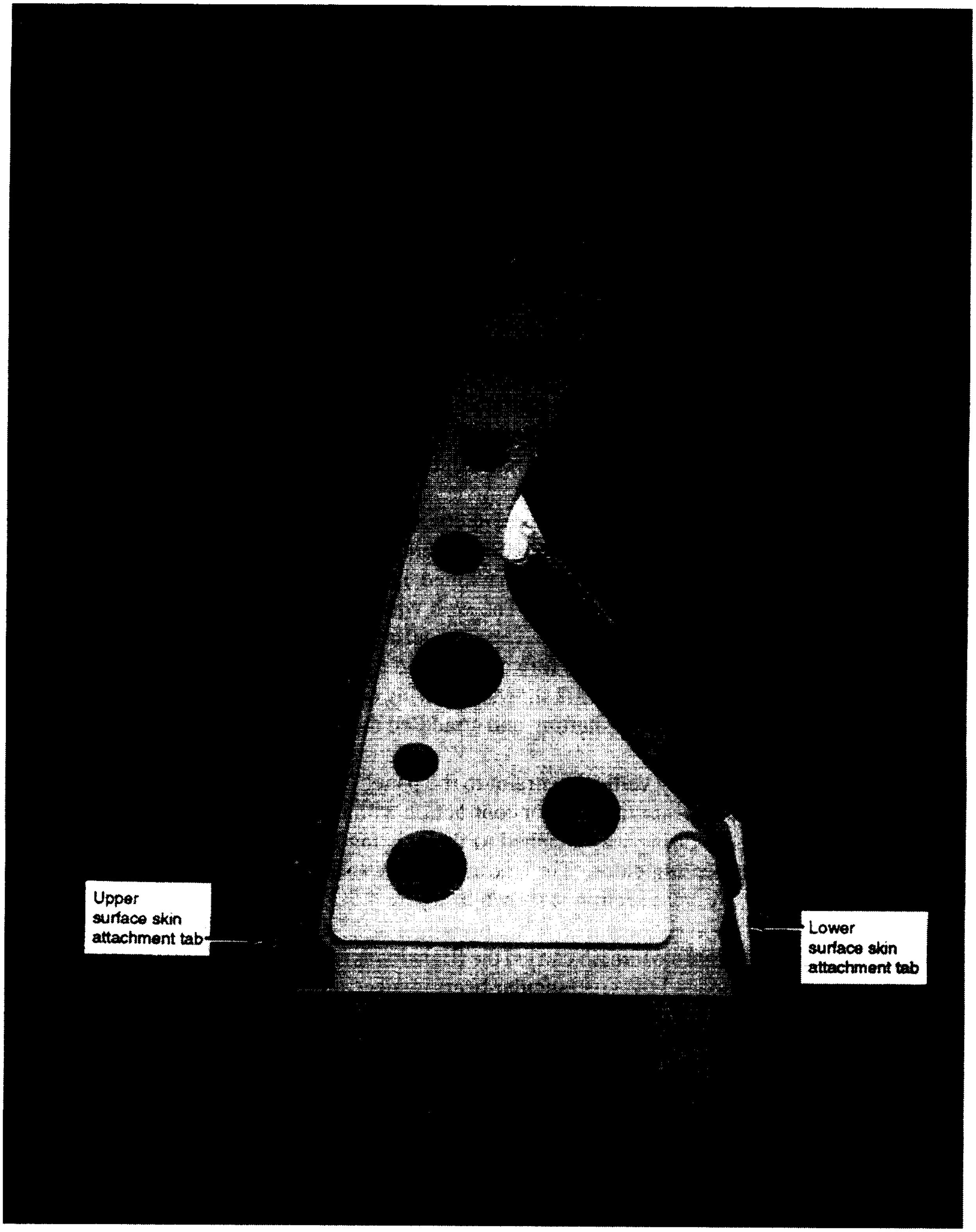

Figure 4.5-1. Typical Airload Rib

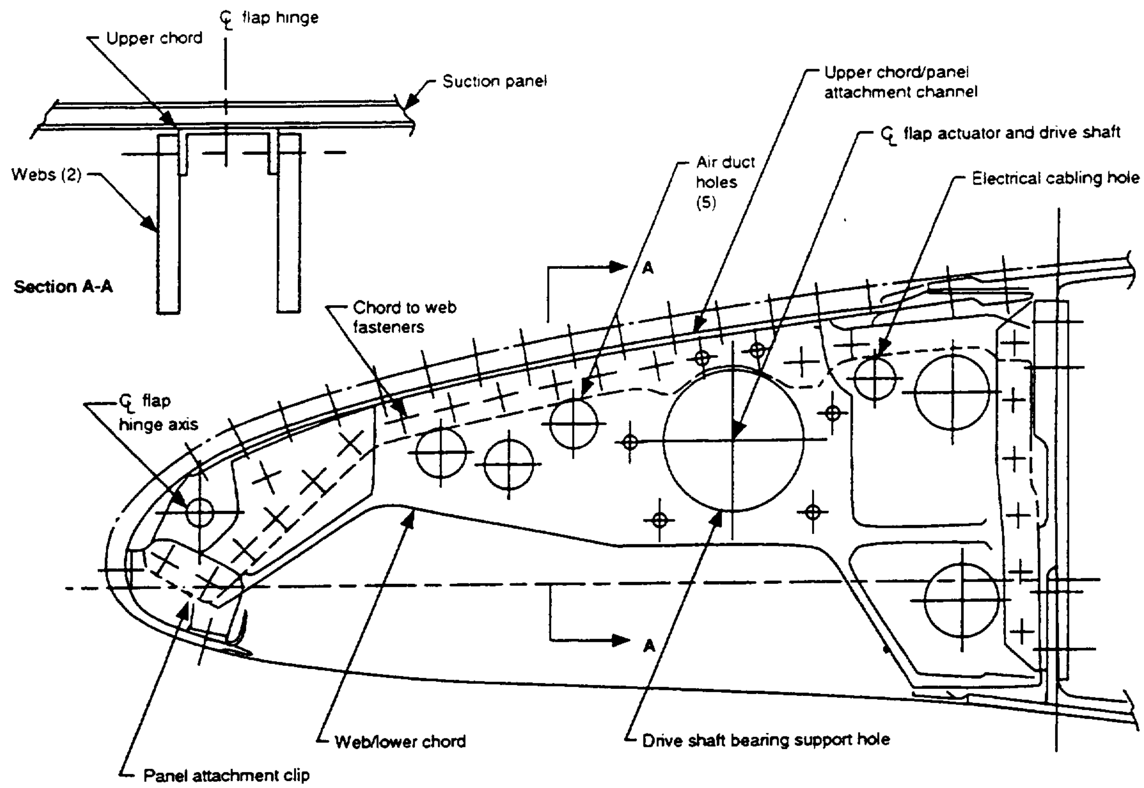

Figure 4.5-2. Typical Hinge Support Rib Assembly

N30677-060M

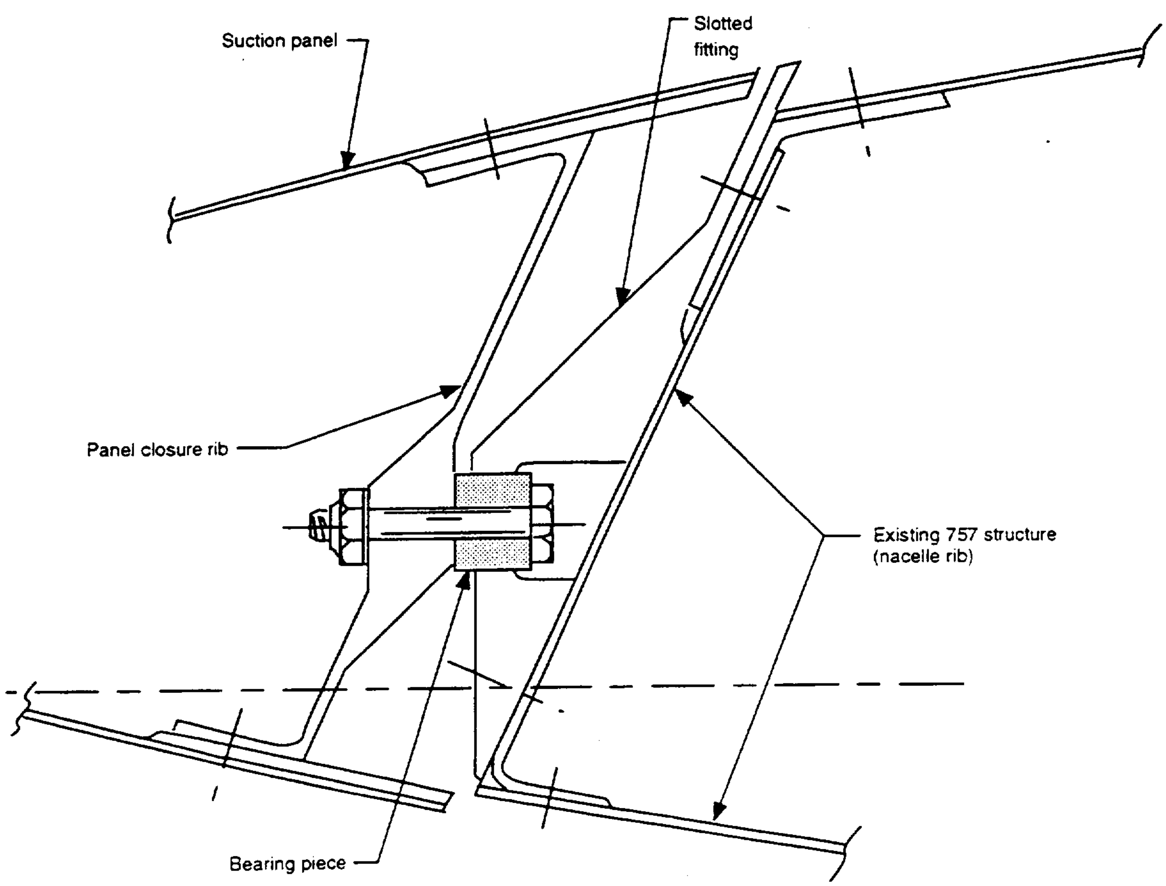

Figure 4.5-3. Section Through Inboard Panel Closure Rib

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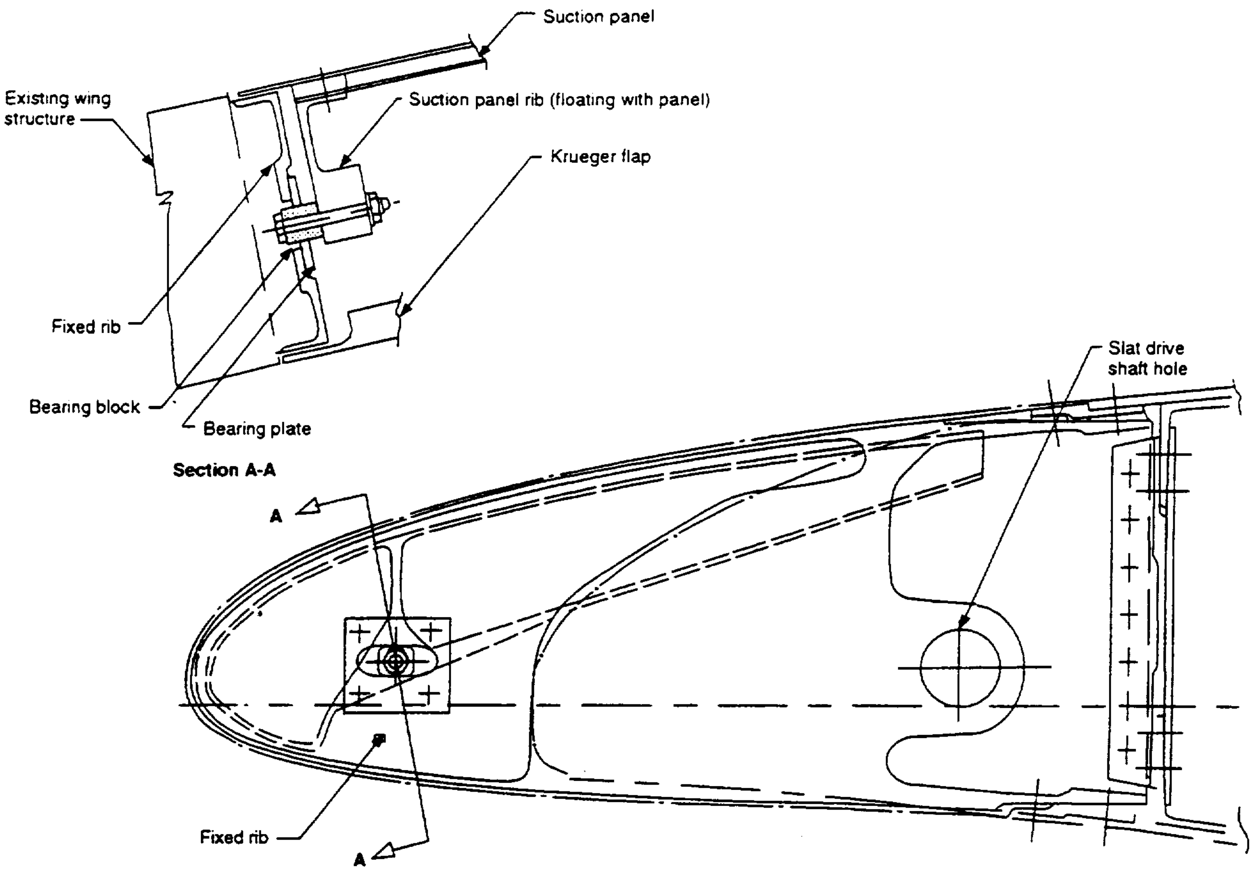

Figure 4.5-4. Outboard Closure Rib

4.6 KRUEGER FLAP AND DRIVE SYSTEM

The Krueger flaps are used both for lift augmentation and as insect shields for the suction panel. Figure 4.6-1 shows cross sections through a typical support location with the flaps in deployed and retracted positions. A simple four-bar linkage translates the main flap through 135 deg of travel. A slave linkage deploys the bullnose during this motion. For good low-speed performance, flap camber is provided by a large radius on the bullnose coupled with built-in curvature in the main flap, obtained by modifying the lower wing leading edge from the original 757 contour. Flap spans are 136.7 in inboard and 132.6 in outboard. The average chord when deployed is 19.3 in.

To achieve the unusually high flap position needed to shield the leading edge from insects, the flap kinematics required a pivot point set high in the upper forward part of the support structure. Attachment-line aerodynamic requirements associated with the location of the stagnation point set

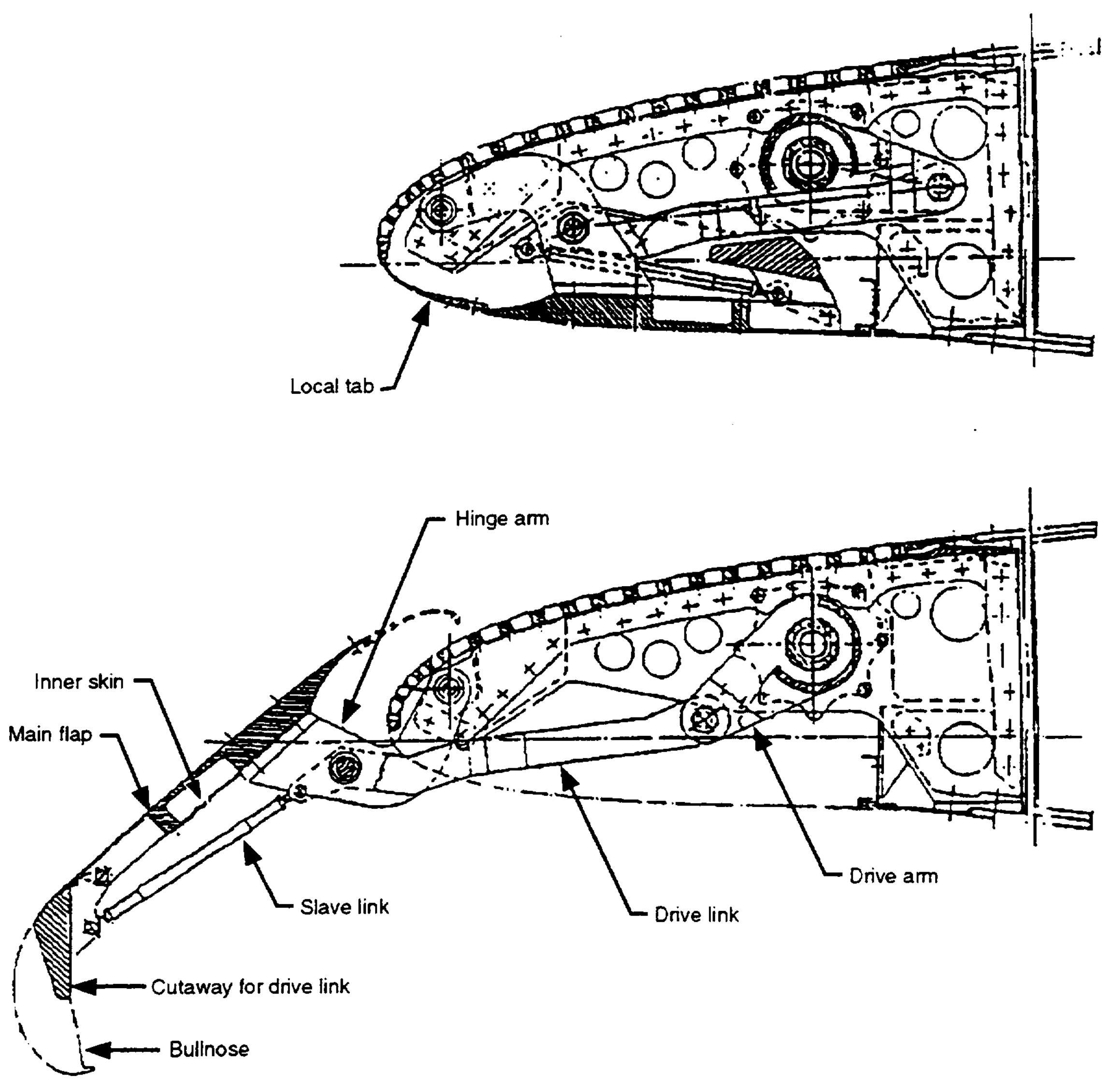

Figure 4.6-1. Cross Sections at Typical Flap Support Rib

critical dimensions for cutouts beneath the highlight. To meet these conditions, the hinge arms were "goosenecked." The cutouts formed in the panel to provide hinge arm clearances when flaps are deployed are closed by local sealing tabs carried on the flap. Operation of this device was described in the section on the suction panel structure and is shown in figure 4.3-9. The main flap components were machined from 7075-T7351 aluminum alloy and have heavy inner pocketing as well as the necessary external contoured surfaces. The requisite clevises, lugs, and bores for the common hinge between the main flap and the bullnose were also machined into the details. To provide spanwise and torsional stiffness, the main flap has an inner skin of 7075-T6 aluminum sheet bonded and riveted to the basic member. The bullnose is slotted to provide clearance at each drive link location.

The drive link is "dog legged" to minimize the slot in the bullnose and straddles the hinge arm at the forward end, providing clevises for the two slave links that drive the bullnose. A clevis is provided in the drive arm from the rotary actuator for the drive link lug. The hinge, drive link, and drive arm were machined from heat-treated high-strength steel.

The Krueger flaps were designed to operate in conjunction with, and be driven by, the existing slat drive system (fig. 4.6-2). The drive line is intercepted at the nacelle and is connected to an offset gearbox (ratio 1.14:1) to provide the needed rpm for the 747-type rotary actuators that were the final drive to the flap linkage. The shafting passes the full length of the flap bay and is returned to the existing slat drive line by an offset gearbox ratioed back to the slat drive rpm. The offset gearboxes also align the shafting with the rotary Krueger actuator location needed to operate the flaps. Figure 4.6-3 shows actuator positioning and a typical cross section through the drive arm, drive arm support, and shafting across the linkage. Flaps are positioned and timed using the existing slat control devices. The "slat sealed" (takeoff) position was deprogrammed to allow full-cycle operation of the Krueger flaps.

To maintain flap and slat position, both extended and retracted locking latches were installed. These devices rotationally lock the drive shaft at the input and output ends of the test section and are manually controlled by the pilot.

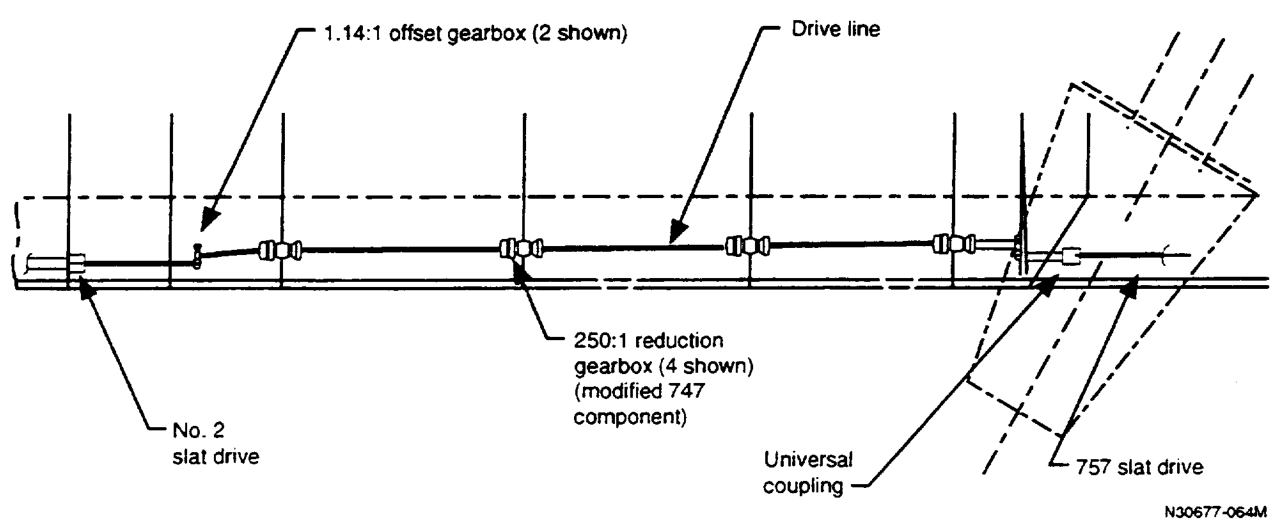

Figure 4.6-2. Modified Slat Drive System

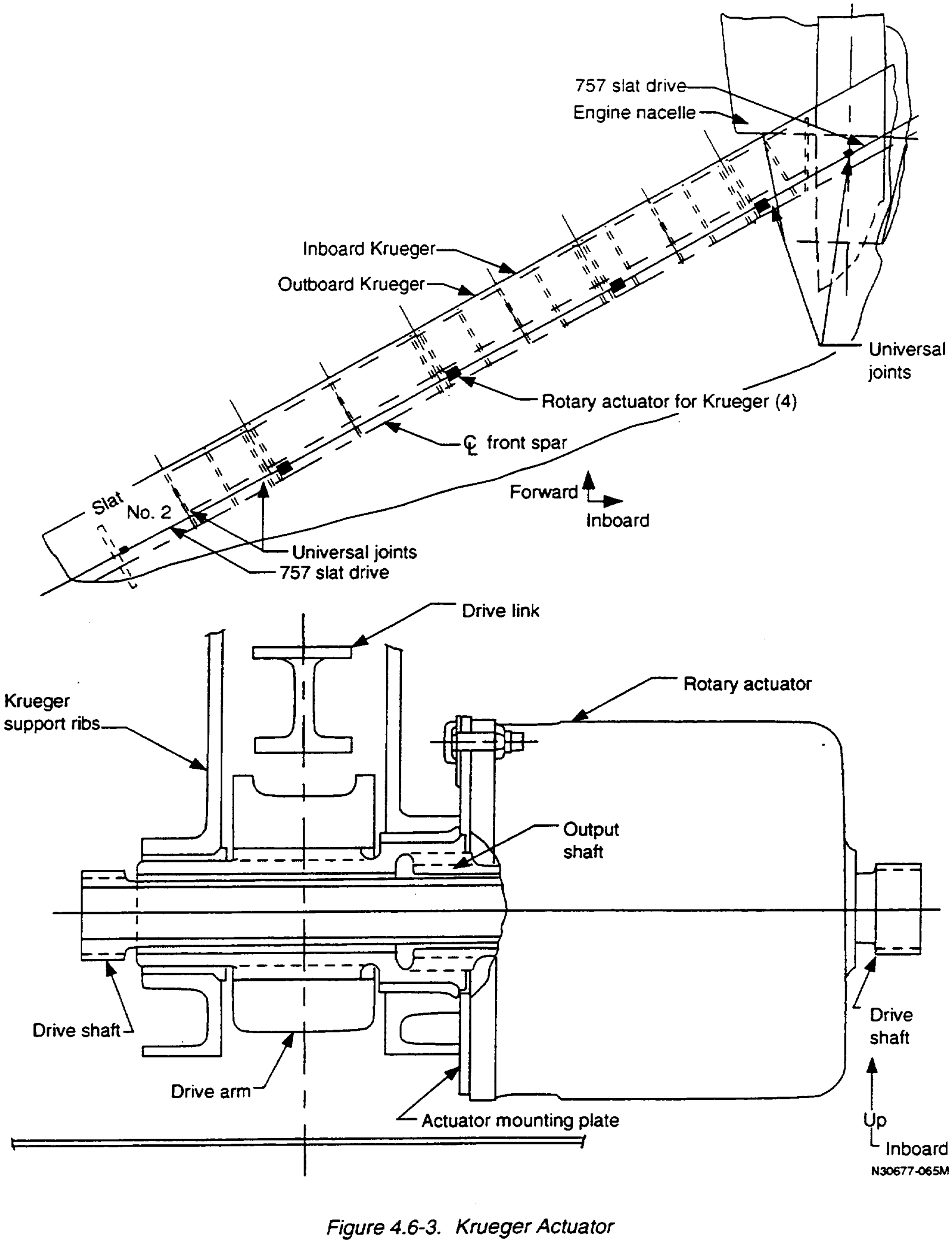

Figure 4.6-3. Krueger Actuator

4.7 LEADING EDGE INSTALLATION

Installation of the leading edge was complicated by three considerations:

a. The attitude of the leading edge with respect to the wing had to be accurate to within a fraction of a degree to ensure that the pressure distribution in flight would meet the HLFC requirement.

b. Both the entire panel and the joint had to meet the aerodynamic smoothness requirements established in volume II and shown in figure 3.2-1 of this volume.

c. The possibility of damage during the numerous trial fittings that were anticipated had to be minimized. Any appreciable damage would not only be costly but could lead to a delay of a year or more in the experiment.

To deal with these problems, a fixture was constructed for holding the flexible leading edge in the proper shape and position during installation. Figure 4.7-1 shows the fixture positioned ahead of the leading edge. Figure 4.7-2 is an isometric drawing of the installation fixture.

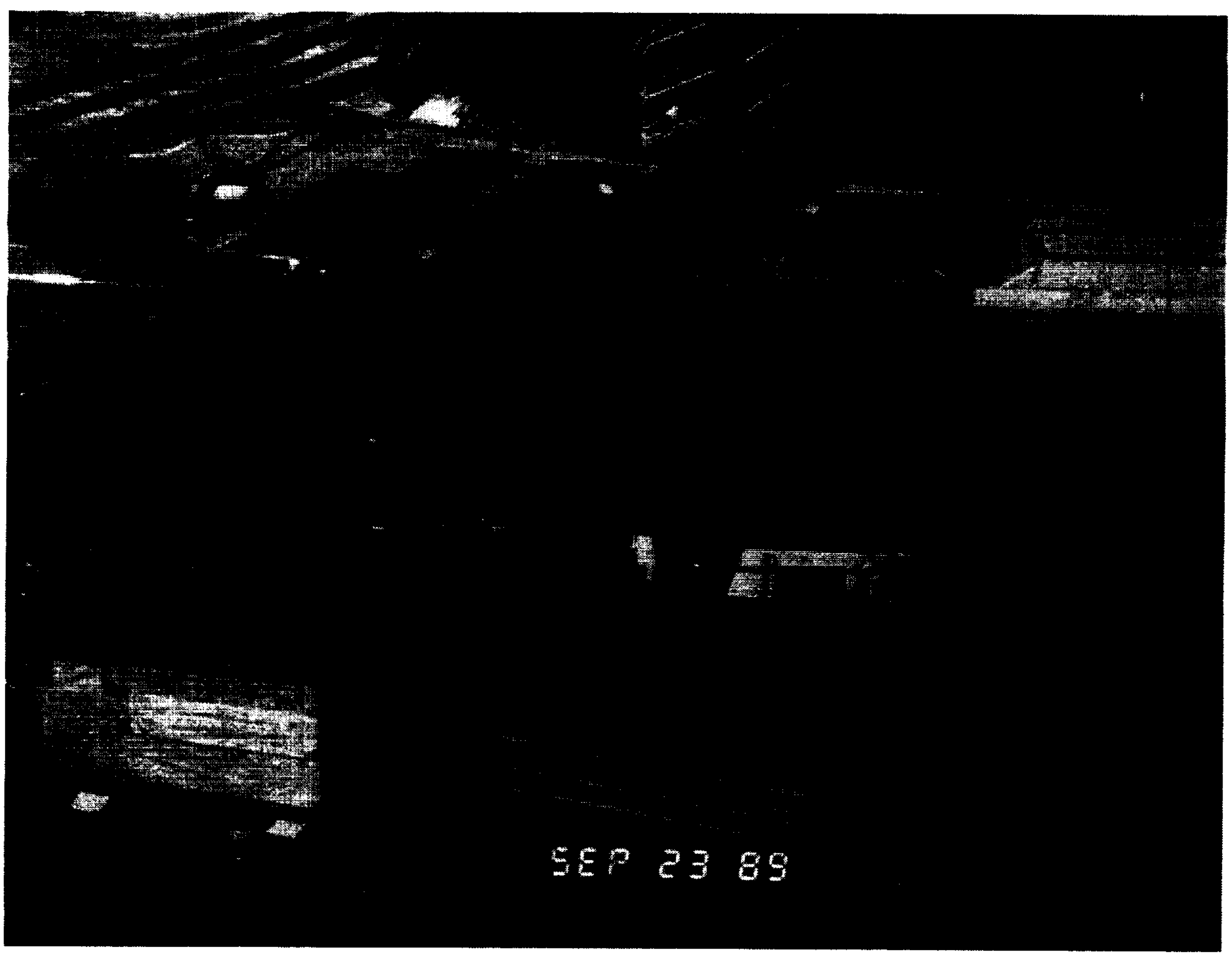

Figure 4.7-1. Installation Fixture

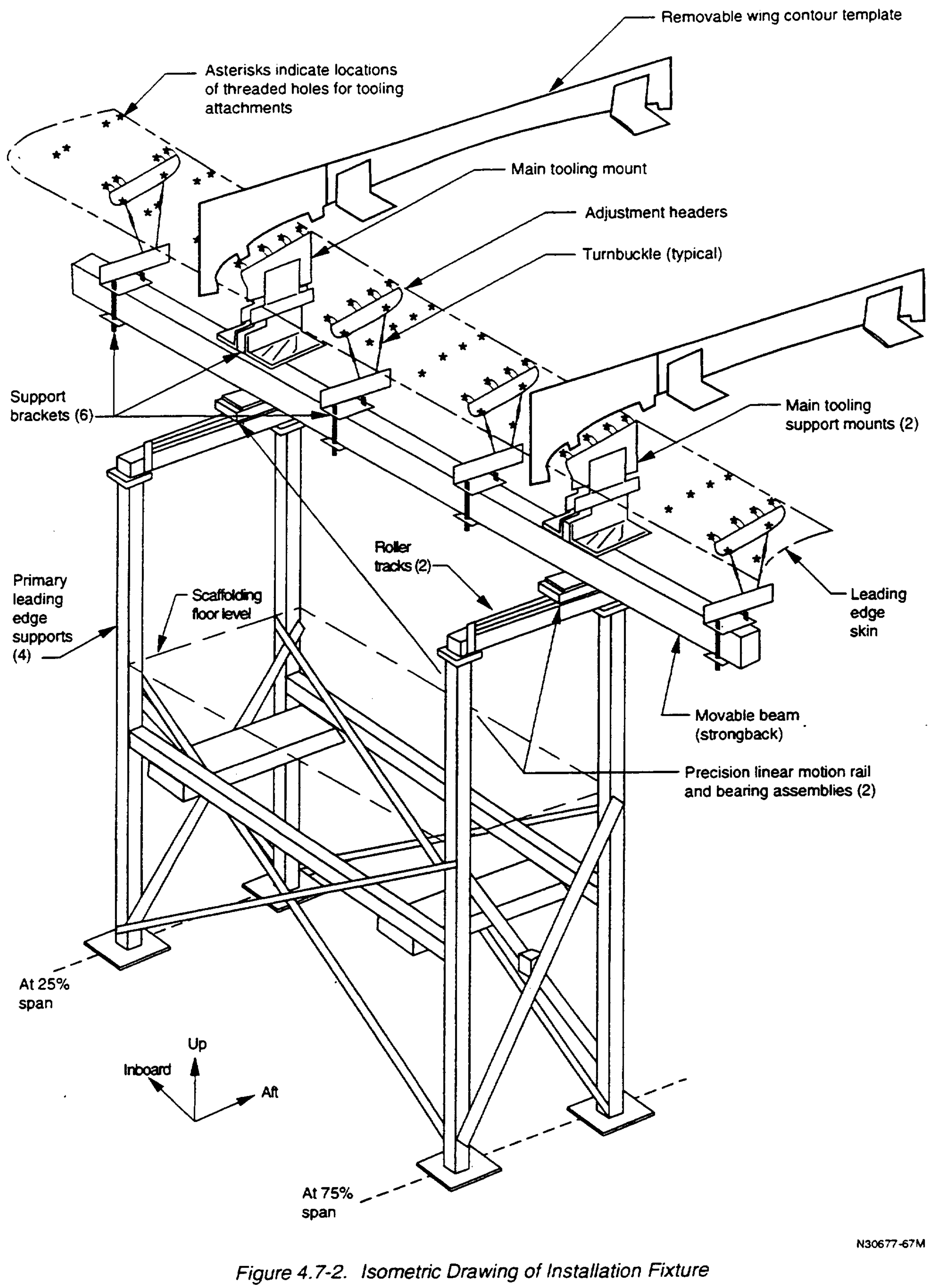

Figure 4.7-2. Isometric Drawing of Installation Fixture

To solve the problems of position and shape, the leading edge was supported by a rigid beam (strongback) to which a number of support brackets were affixed. The strongback was also supported in two places on roller tracks that allowed the beam and leading edge to move freely in the fore and aft direction. These primary leading-edge supports were at approximately 25% and 75% of the leading-edge span. The supports were also provided with precision linear motion rail and bearing assemblies to allow for inboard-outboard positioning of the leading edge, as well as height and angular adjustments. Between the two main supports, several adjustment headers fitted with turnbuckles allowed the leading edge to be restrained in its correct shape while it remained supported on the strongback. Figure 4.7-3 shows a view of the leading edge, rigid beam, and fixture details from below.

The fixture reduced the possibility of damage because the assembly could be moved into the proper position without the risk of dropping or twisting the leading edge. Figure 4.7-4 shows the wing contour templates used to ensure correct alignment of the leading edge. These templates were designed from the aerodynamic lofts of the wing and attached to it at the front and rear spars. The templates were designed for use with gauge blocks and do not touch the wing, eliminating another potential source of damage. The front one-third of the template was made removable so that the

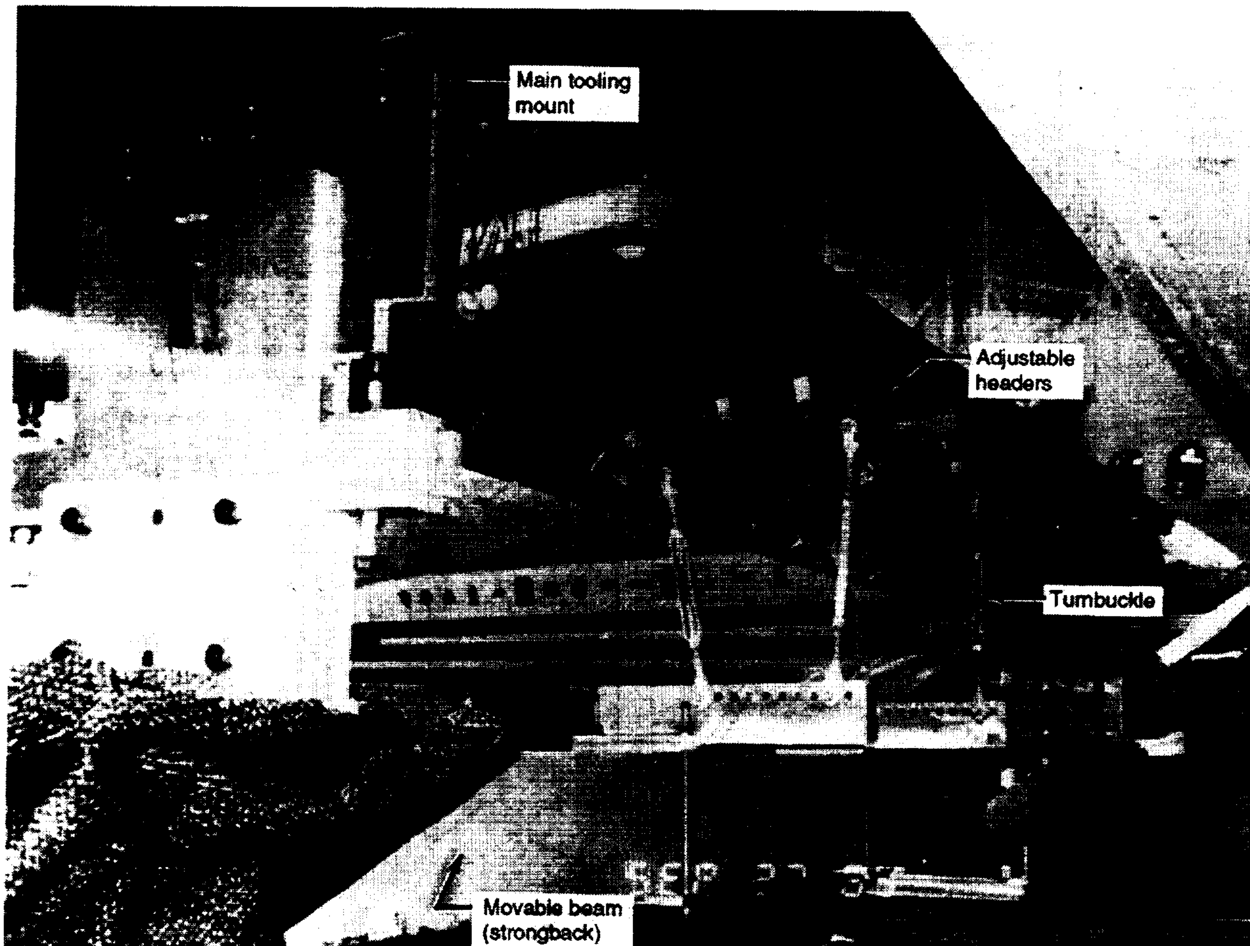

Figure 4.7-3. Underside View of Leading Edge and Installation Fixture

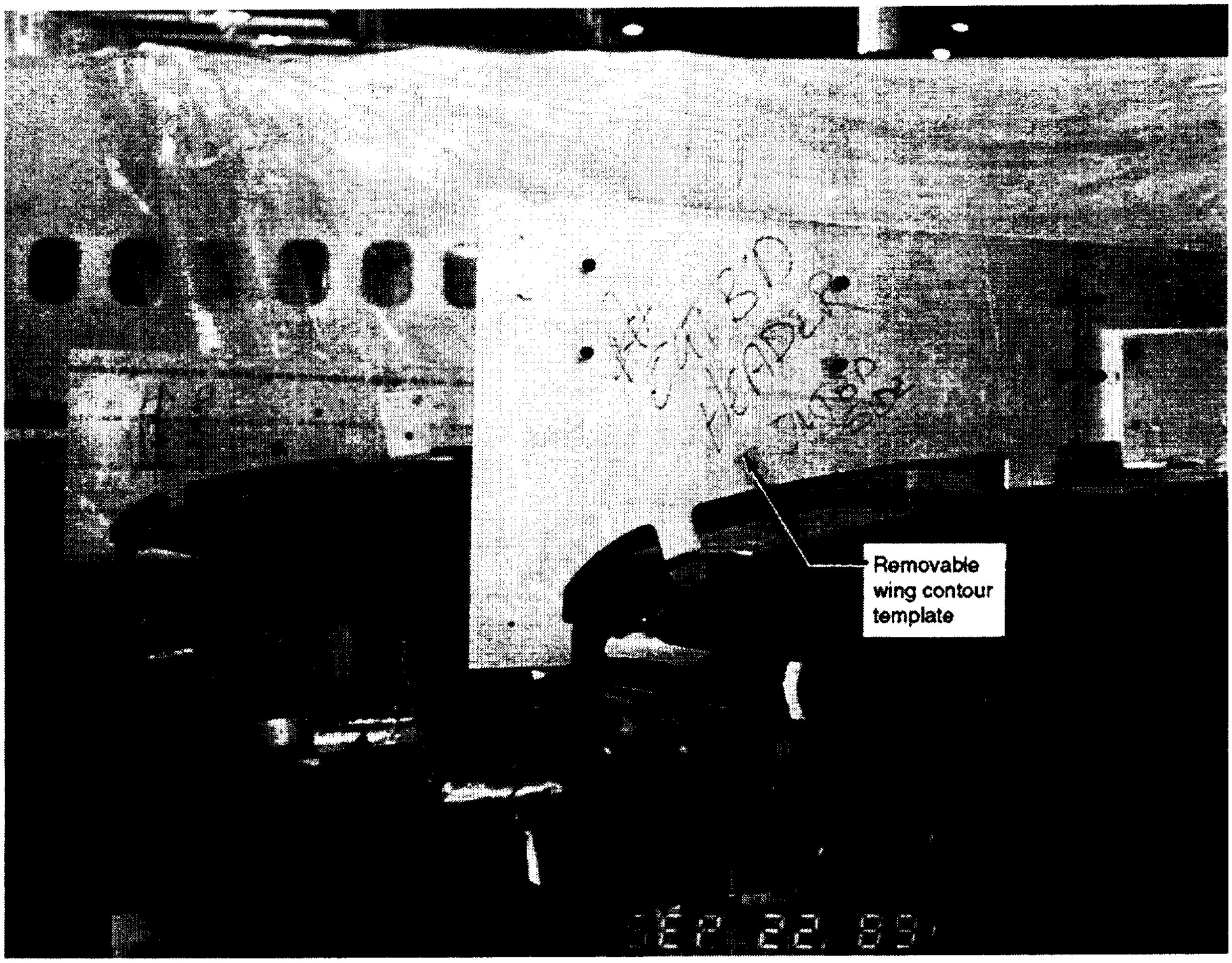

Figure 4.7-4. Alignment Templates

attachments of the templates to the wing would not be disturbed when moving the leading edge away from the front spar.

Cross sections through the panel to front spar joint are shown in figures 4.7-5 through 4.7-7. The suction panel was attached to the wing surface by a continuous splice plate shown in figure 4.7-6. This plate is part of the existing design where it is used to support the upper surface panels of the fixed leading edge. Its shape and hole pattern dictated the design of the panel closeout stringer shown in figure 4.7-5, where the under surface was revised to match the splice plate. To minimize surface slope changes at the joint, the outer surface of the stringer was machined to suit the local wing contour.

Before assembling the panel to the wing, waviness surveys of the surface were made with the portable gauge shown in figure 4.7-8. These surveys showed that a band adjacent to the front spar would require some rework to meet the surface criteria. The rework consisted of filling the joint between the HLFC leading-edge panel and the existing wing structure, filling a spanwise wing skin splice, and filling and smoothing low spots, fasteners, and irregularities. The surveys also showed that the wing surface could be used to set panel height and eliminate or minimize discontinuities during assembly.

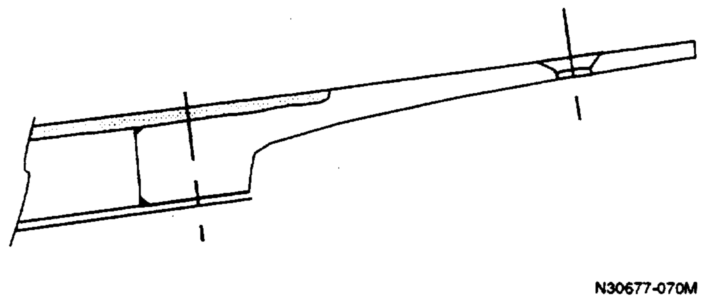

Figure 4.7-5. Suction Panel Closure Stringer

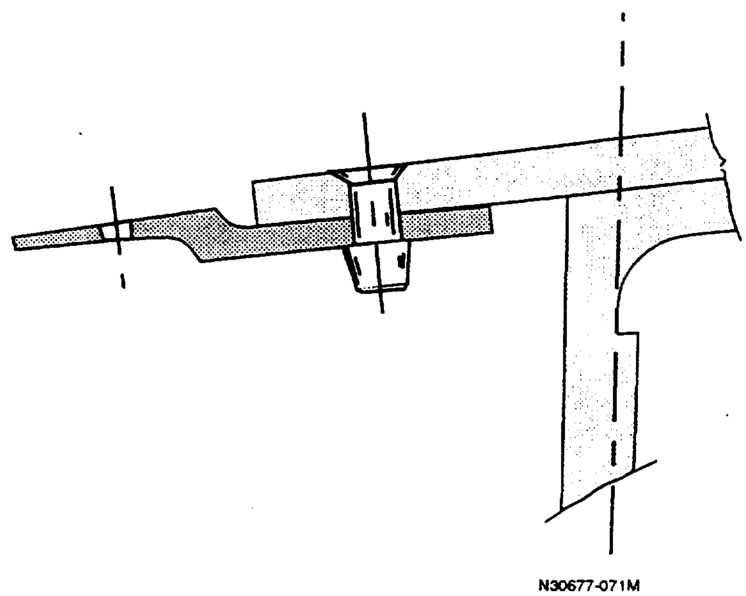

Figure 4.7-6. Wing Interface Structure

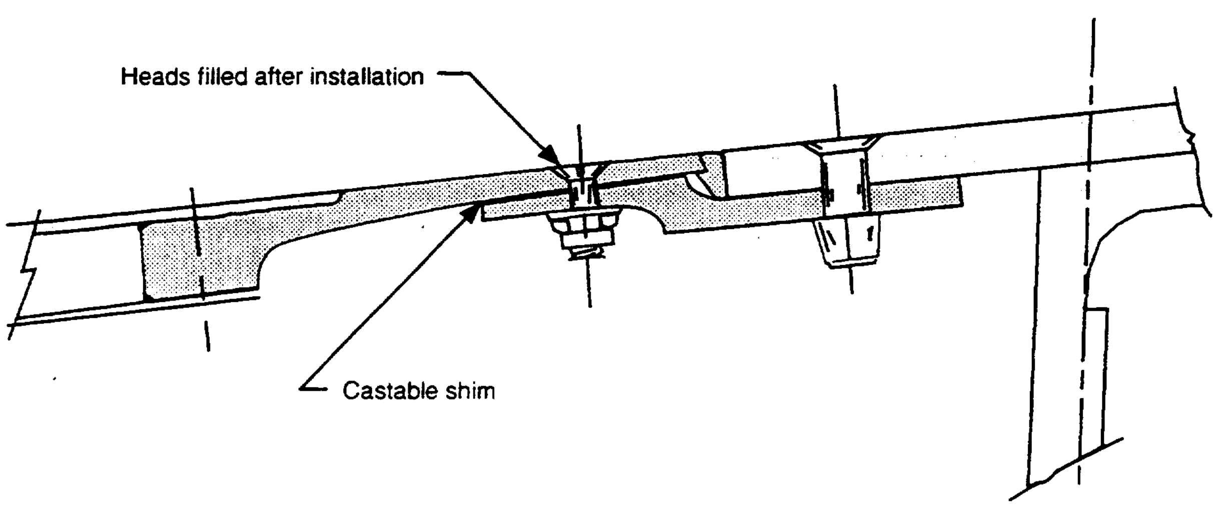

Figure 4.7-7. Completed Joint Panel to Front Spar

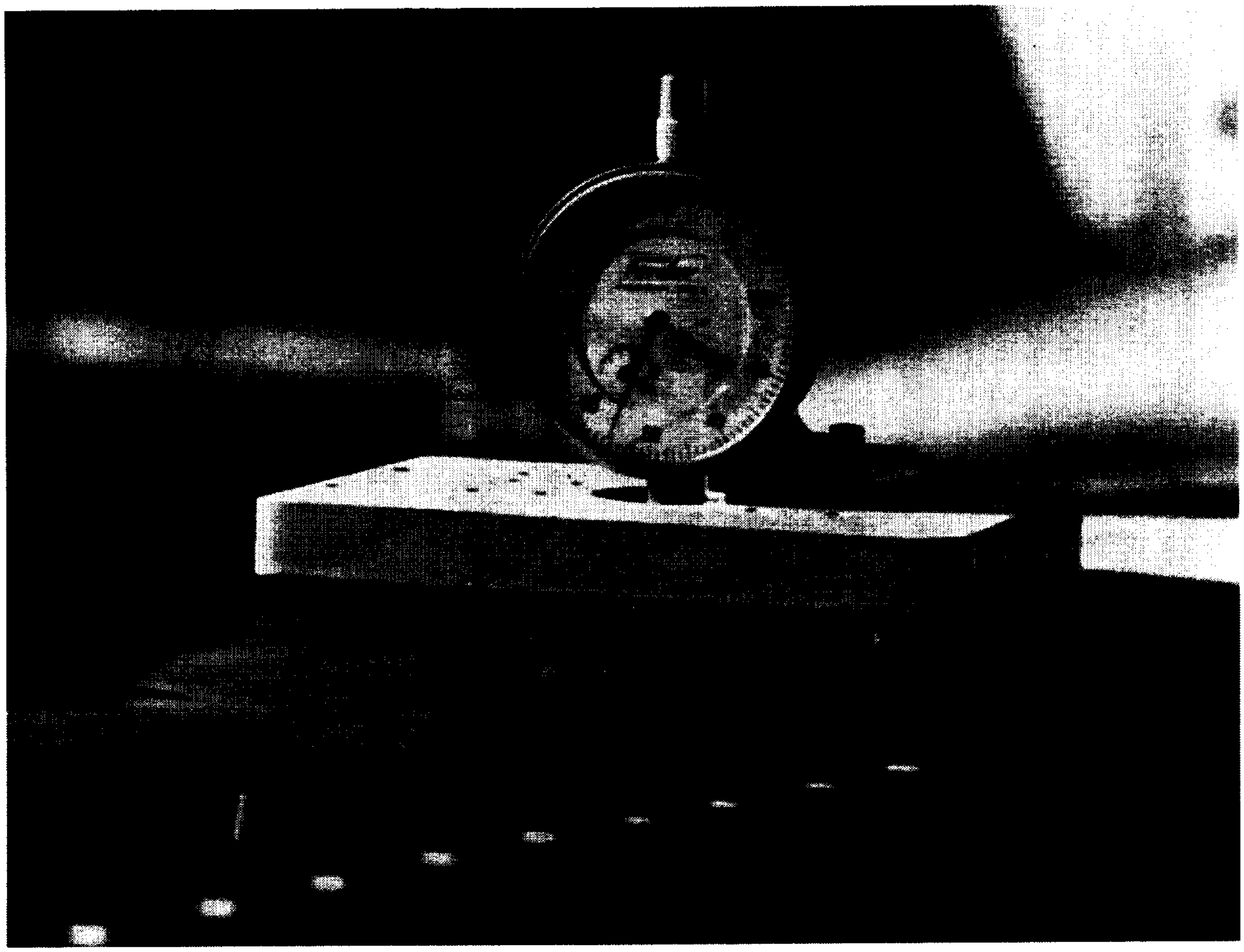

Figure 4.7-8. Manual Waviness Gauge

A further survey of the splice plate mating surface indicated a need for some form of variable shimming. The joint between the plate and the panel was then redesigned to allow the use of a gap-filling castable shim during final assembly.

A procedure was developed using a series of 31 gauge blocks that provided surface indexing from the existing wing surface and at the same time effectively clamped the panel edge into a final faired position. The gauge blocks are shown in position in figure 4.7-9. This clamping arrangement allowed for the application of the castable shim material at the appropriate time. After installation of the clamps, a further survey confirmed an acceptable fit, and the assembly was programmed for the castable shim application.

A release agent applied to the underside of the panel allowed for disassembly, if required, to facilitate installation of the panel. The castable shim material was applied, and the joint reassembled finger tight, allowing the excess shim to ease out before curing. Following the cure, clamps were removed, the final holes were drilled from the splice plate and countersunk, and close-tolerance high-strength shallow-head bolts were installed to complete the joint.

Figure 4.7-9. Gauge Blocks

The wing upper surface and panel joint were painted and filled to achieve laminar flow smoothness requirements. The materials used for this and their properties are described in appendix B.

The wing was then checked for contour, and waviness was surveyed at 13 span stations. Results of the survey are shown in figure 4.7-10. The surface met the waviness criterion (less than 0.002 in over any 2-in chordwise distance) at all but three locations, which were just ahead of the front spar joint, at WBL 347, 359, and 455. (The line rises sharply at the leading edge for all stations. This is not waviness but the curvature of the airfoil contour.) It did not appear feasible to improve the surface further, so the installation was accepted.

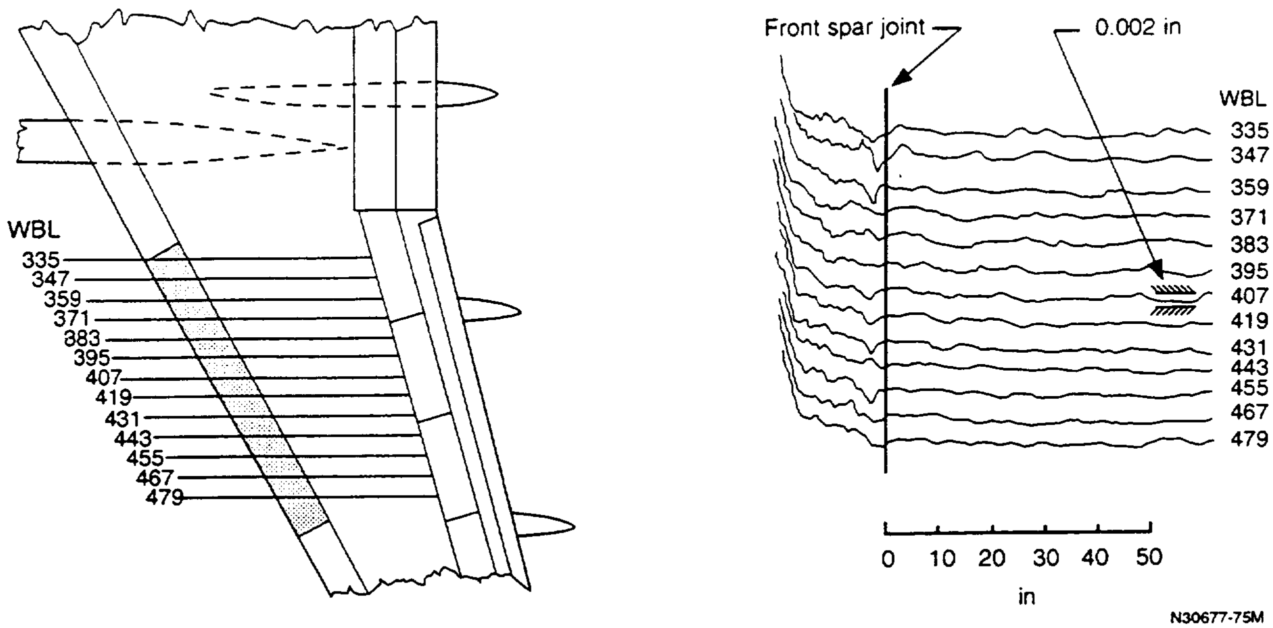

Figure 4.7-10. Waviness Survey Results

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5.0 STRUT FAIRING

The suction compressor, oil cooler, anti-ice heat exchanger, and various other components were located in the engine strut, making it necessary to develop a new fairing to enclose the new components. The fairing was constructed with conventional experimental airplane techniques using stretch-formed skins of 0.15-in-thick aluminum and mechanical fasteners. Figures 5-1 and 5-2 show the strut as installed on the airplane.

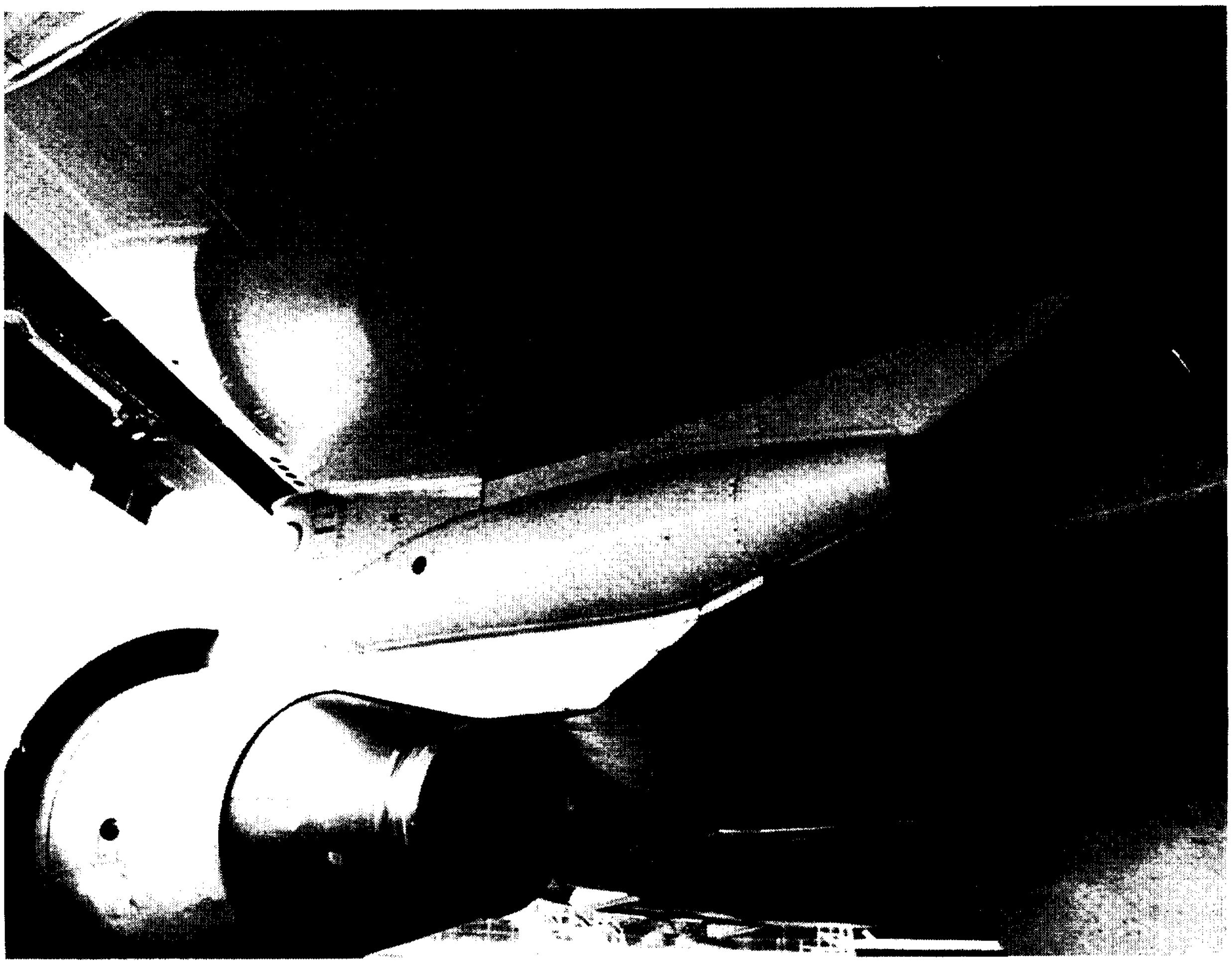
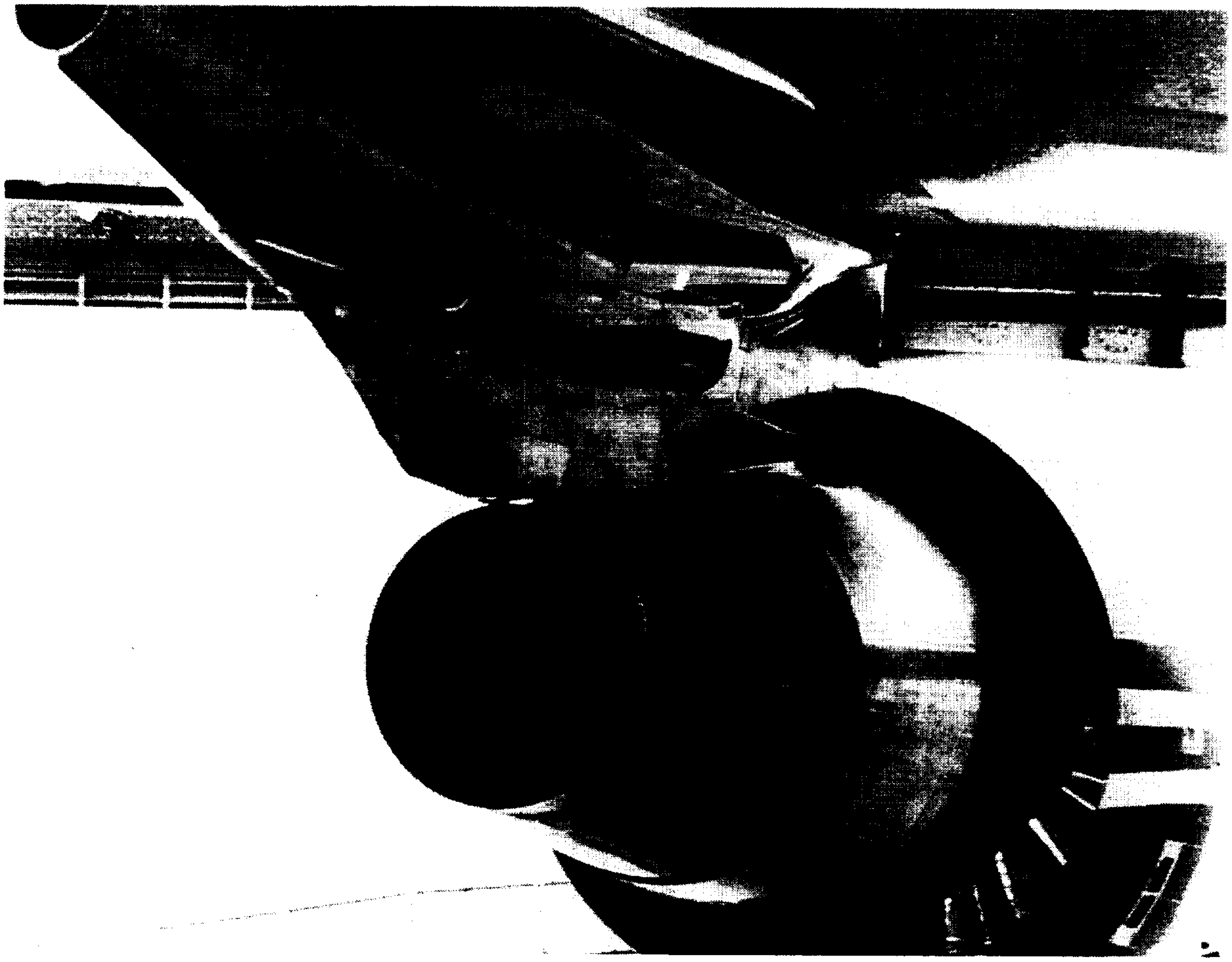


Figure 5-1. Modified Strut (Outboard Side) N30677-76M



N30677-77M

Figure 5-2. Modified Strut (Inboard Side)

6.0 REFERENCES

1. Northrop Corporation, *Final Report on LFC Aircraft Design Data, Laminar Flow Control Demonstration Program*, Northrop Report NOR 67-136, June 1967.

2. Fisher, D. F., and M. C. Fischer, *The Development Flight Tests of the JetStar LFC Leading Edge Flight Test Experiment*, NASA CP-2487, 1987.

3. Wagner, R. D., D. V. Maddalon, D. W. Bartlett, F. S. Collier, and A. L. Braslow, *Laminar Flow Flight Experiments*, NASA CP-3020, Transonic Aerodynamics Symposium, NASA Langley Research Center, 1988.

4. Maddalon, D. V., and A. L. Braslow, *Simulated-Airline-Service Flight Tests of Laminar Flow Control With a Perforated Surface Suction System*, NASA TP2966, March 1990.

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APPENDIX A

ADHESIVE BONDING DEVELOPMENT

A.1 BACKGROUND

The HLFC suction panel (see fig. 4.3-1) was required to meet precise contour and waviness requirements, as presented in volume II. This precluded mechanical attachment methods since fastener heads are roughness elements and tight clearance interference fits cause local skin bulging distortion. Therefore, structural adhesive bonding was chosen as the attachment process.

A.2 REQUIREMENTS

Skin porosity and thermal resistance were the two basic requirements that determined the selection of adhesive materials and processes. Because the outer skin porosity was of primary importance, care was taken to preserve hole size during surface preparation and priming and to control adhesive flow or squeeze-out during bonding. The design of the HLFC system called for a thermal anti-icing (TAI) system using engine bleed air. The original adhesive temperature requirement was 310°F. It was later revised to 400°F for safety in the event of TAI system failure.

Skin porosity was checked at several stages in the process by measuring the flow through small areas of the skin (typically 4 by 1 in) at a fixed differential pressure (typically 100 psf). Figure 4.4-4 shows such a measurement in progress on the 4-ft verification panel.

A.3 RESULTS

A.3.1 Surface Preparation

The only titanium surface preparation process for which tanks large enough for the 3- by 22-ft HLFC suction panel were available was the phosphate-fluoride conversion coating process currently used in production of CH-47 helicopter blade caps. Some modifications to the process were required to protect the laser perforations on the outer skin.

Short-duration nitric-fluoride acid etching had been found to have little effect on the porosity of perforated sheets used in another program. However, when smaller diameter laser perforations such as those of the HLFC suction panel were tested, significant porosity increases were noted. Hole size was preserved by masking the outer (nonbond) surface of the perforated sheet with a mylar adhesive tape. Masked sheets showed no significant increases in flow due to etching.

During production of the final suction panel, unacceptable reductions in flow were observed after the surface preparation process. To isolate the problem, a subsize (4 ft) perforated panel was put through the process. It was found that particles of residue from the surface preparation process caused hole blockage. The panel was then washed from the outer (nonbond) side to avoid disrupting the conversion coating. This was found to restore acceptable flow characteristics.

A.3.2 Priming

Adhesive primer applied to the perforated titanium sheet clogged holes, causing excessive flow reduction. This was eliminated by applying a small positive air pressure during primer application. The opposite side of the panel was vacuum bagged and attached to shop air, resulting in sufficient flow to inhibit clogging, as indicated by flow testing.

A.3.3 Adhesive Evaluation

Adhesive testing was conducted in three phases. Phase one consisted of system screening and manufacturing evaluations. Phase two included property evaluation and maximum temperature determination. Phase three included verification of mechanical properties.

Mechanical properties were assessed using single lap shear testing and stringer tension tests.

A.3.4 Adhesive Screening

Nitrile-phenolic structural adhesive systems were chosen for initial testing on the basis of positive experience by McDonnell Douglas on earlier laminar flow hardware and by Boeing on titanium helicopter blade caps. Unsupported 10-mil 3M AF-30 and AF-31 nitrile-phenolic systems were subjected to lap shear tests over a temperature range of -65° to 450°F. Lap shear tests were also done in -65°F, room temperature, and 400°F thermal cycles. AF-30 was found to have inadequate strength at 260°F for this application. AF-31 had reasonable lap shear strength retention to 350°F. Increased strength at 260°F after aging and thermal cycling indicated that temperature performance could be improved by postcuring.

These adhesives were also tested on small panels of similar design to that expected for the panel. Manufacturing evaluations indicated that AF-31 did not have adequate tack or squeeze-out control. Lack of tack was expected to cause a serious locating problem where the bond line was nearly vertical, such as at the highlight.

Alternate adhesives were evaluated to address the manufacturing concerns. A thinner (3-mil) version of 3M AF-31, AF-15, and a high-temperature epoxy (B.F. Goodrich PL780) were evaluated. AF-15 was too thin to provide a consistent, void-free bond line. PL780 was chosen because it showed adequate properties and better process characteristics. Poor property retention at 350°F indicated that high-temperature performance would be sacrificed for process ability. A summary of adhesive systems that were screened and their attributes is shown in table A-1.

A.3.5 Property Evaluation

Stringer tension tests were conducted at 230°, 260°, and 290°F to determine the maximum usable temperature of the PL780/PL502 adhesive/primer system. On the basis of the test results, shown in table A-2, the bond line temperature was limited to 240°F.

A.3.6 Property Verification

Thirty lap shear and stringer tests were conducted at 240°F and ten lap shear and stringer tension tests were conducted at room temperature to establish design allowables. Test results are shown in figure A-1.

Table A-1. Structural Adhesives

Designation	Supplier	Thickness	Primer	Chemistry	Advantages
AF-30	3M	10 mil, unsupported	EC1660	Nitrile-phenolic	Low flow, production experience
AF-31	3M	10 mil, unsupported	EC2174 or EC1660	Nitrile-phenolic	Low flow, high-temperature properties
AF-15	3M	3 mil, unsupported	EC2174 or EC1660	Nitrile-phenolic (same as AF-31)	Low flow, high-temperature properties, squeeze-out control
PL780	B. F. Goodrich	7 mil, unsupported	PL502	Modified epoxy	Better tack, thickness/ squeeze-out control

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Table A-2. Adhesives Test Results

Stringer Tension Data Summary					
Test temp., °F	Number of samples	Mean initiation stress, psi	Standard deviation	Mean peak stress, psi	Standard deviation
230	10	325	43.1	571	42.2
260	9	295	39.3	445	55.0
290	12	153	38.0	228	61.2

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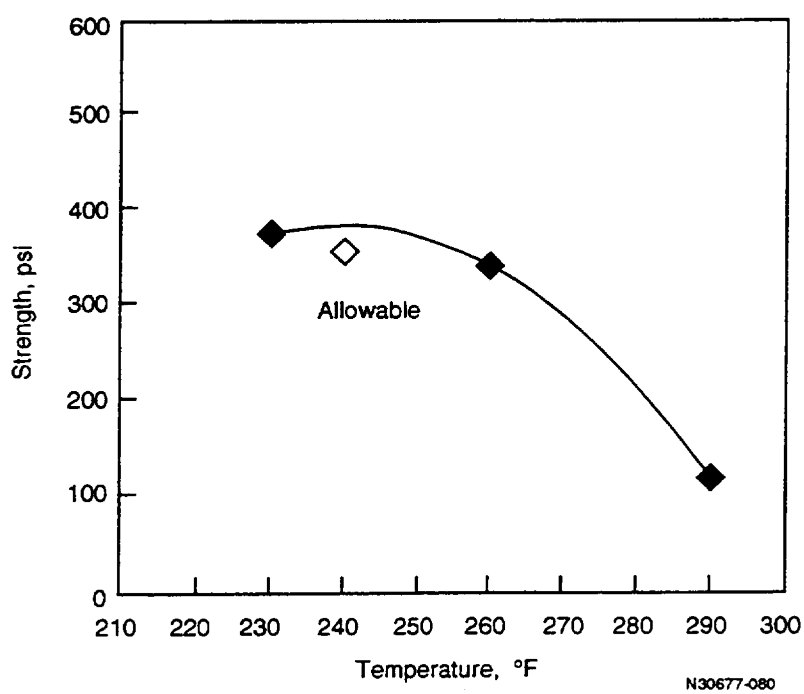

Figure A-1. Adhesive Allowables

These tests show loss of strength as temperature is increased. At 240°F the bond strength is still above the allowable.

A.4 CONCLUSIONS

Based on test results and manufacturing evaluations, a bonding process was developed to fabricate the HLFC suction panel. Features of this process include–

a. Modified phosphate-fluoride conversion coating surface preparation:

   1. The perforated sheet outer surface was masked during nitric-fluoride acid etch to protect holes.

   2. High-pressure wash was added after phosphate fluoride coating to remove residue from holes.

b. Application of positive pressure to perforated sheet to avoid hole clogging during adhesive priming.

c. Selection of the B. F. Goodrich PL780/PL502 adhesive primer system primarily because of processing considerations, such as adequate tack and squeeze-out control. Bond line temperature was limited to 240°F during thermal anti-icing.

APPENDIX B

FAIRING MATERIAL DEVELOPMENT

B.1 INTRODUCTION

The installation of the HLFC experiment modification on the flight test 757 required rework of the wing upper surface aft of the front spar to meet the stringent waviness requirements defined in reference 1. The wing rework consisted of filling the joint between the HLFC leading-edge panel and the existing wing structure; filling a spanwise wing skin splice; and filling and smoothing low spots, fasteners, and irregularities. A sketch of the area involved is shown in figure B-1.

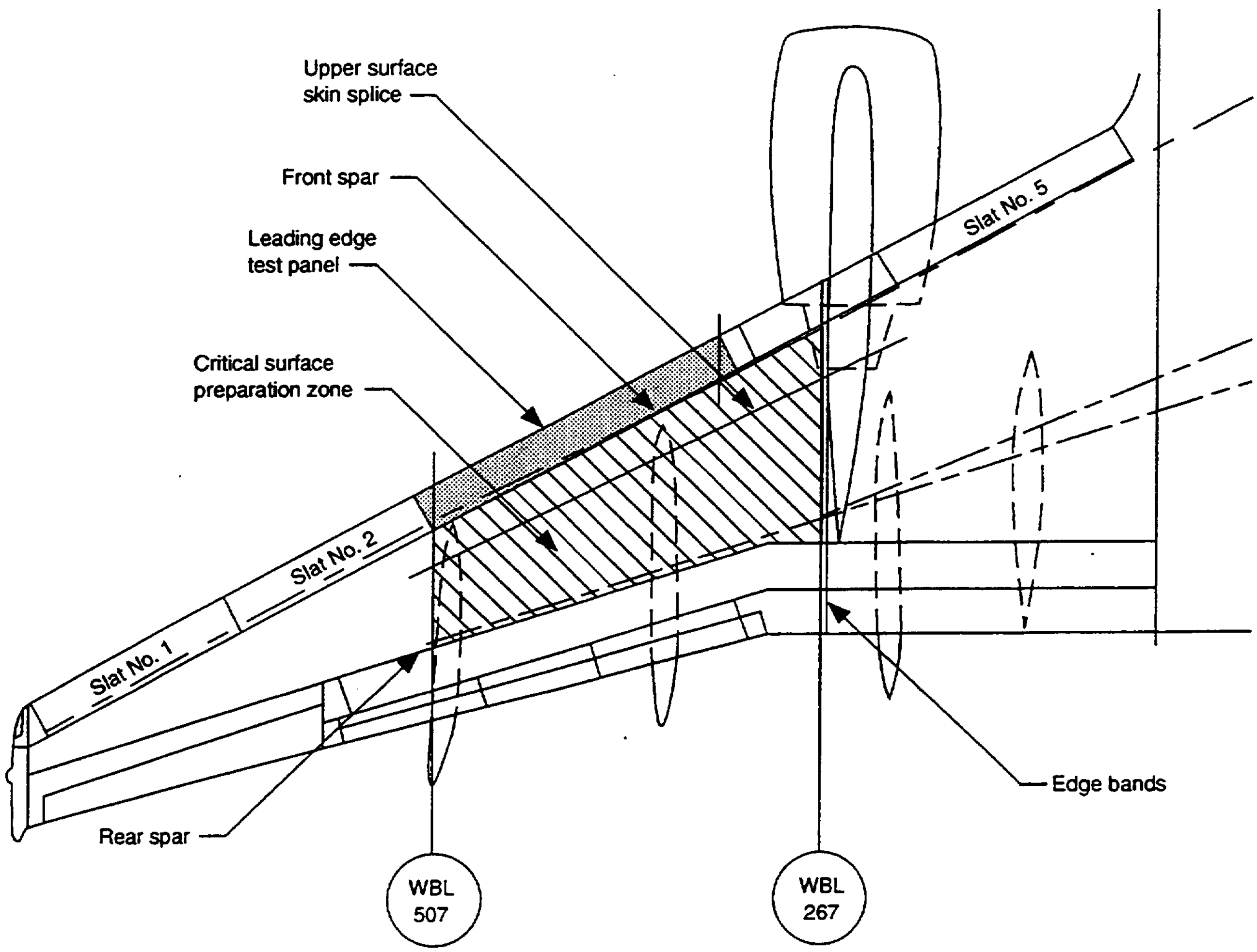

Figure B-1. Wing Surface Rework

Two epoxy filler putties (AKZO 467-9/CA 41B and Sterling U-2706) and two Sprayable epoxy surfacers (AKZO 464-3-1/CA142/TL-52 and Sterling U-2554) were evaluated. The AKZO systems were currently approved for use within Boeing for composite panel finishing. The Sterling systems were recommended by the manufacturer because of their use in similar applications.

To provide a top coat with improved sandability for reworking surface defects, the Sterling urethane enamel top coat was evaluated as an alternative to the very flexible polyurethane enamel currently in use at Boeing. The Sterling systems were selected for final application.

B.2 EXPERIMENTAL METHODS

Adhesion test substrates were constructed of aluminum primed with an epoxy zinc chromate primer. The Sterling and AKZO putties and surfacers were applied to the primed substrate. Sterling putty and surfacer were applied per manufacturer instructions. Specimens were finished by priming and applying either the current Boeing top coat process or Sterling urethane enamel.

Tests included wet/dry tape adhesion, whirling arm (rain erosion), and thermal cycling. Thermal cycle specimens were subjected to 180 to 200 cycles from -65° to 160°F and subsequently examined for evidence of cracking.

Simulated joint specimens were fabricated to evaluate processing methods. A sketch of the simulated joint specimen is shown in figure B-2. This specimen was subjected to 380 thermal cycles and examined for cracks.

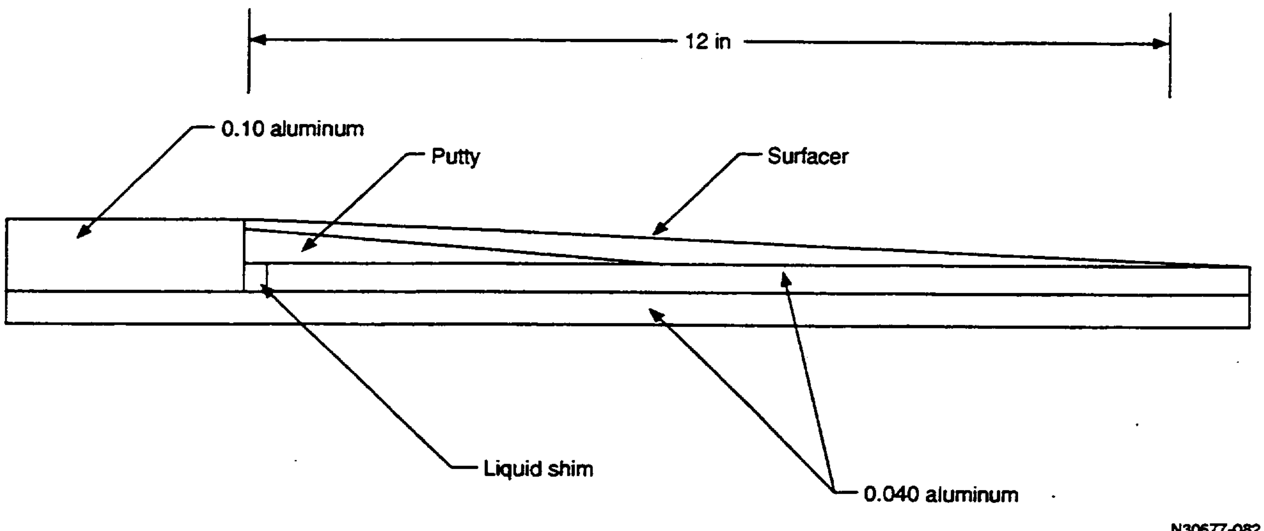

Figure B-2. Simulated Joint Specimen

Joint compression testing was conducted on three simulated joints. Joints were constructed from 2024-T6 aluminum sheet. Each joint gap was filled with two-part, room-temperature-curing epoxy adhesive. Joint gaps and fasteners were coated with materials to be used for fairing of the HLFC panel. One joint was coated with approximately 0.020 in of Sterling U2554 sprayable surfacer, one with 0.030 in of Sterling U2706 putty, and one with putty and surfacer built up to a 0.050-in thickness. These joints were exposed to concurrent mechanical and thermal cycles, as shown in figure B-3. Fifty cycles were applied at compressive load levels of 1.05g and 1.5g and single cycles to 2.0g and 2.5g. Specimens were examined for indications of finish cracking.

B.3 RESULTS

The AKZO putty and surfacer were eliminated because of tape adhesion and rain erosion failures at the filler/finish prime interface. The Sterling surfacer, putty, and top coat passed all adhesion tests and were used on the flight article.

The results of the thermal cycle test indicated that putty thickness must be limited to 0.030 in and surfacer thickness to 0.015 in. Cracking of scribed specimens indicated that discontinuities and stress concentrations should be avoided, particularly in thick applications.

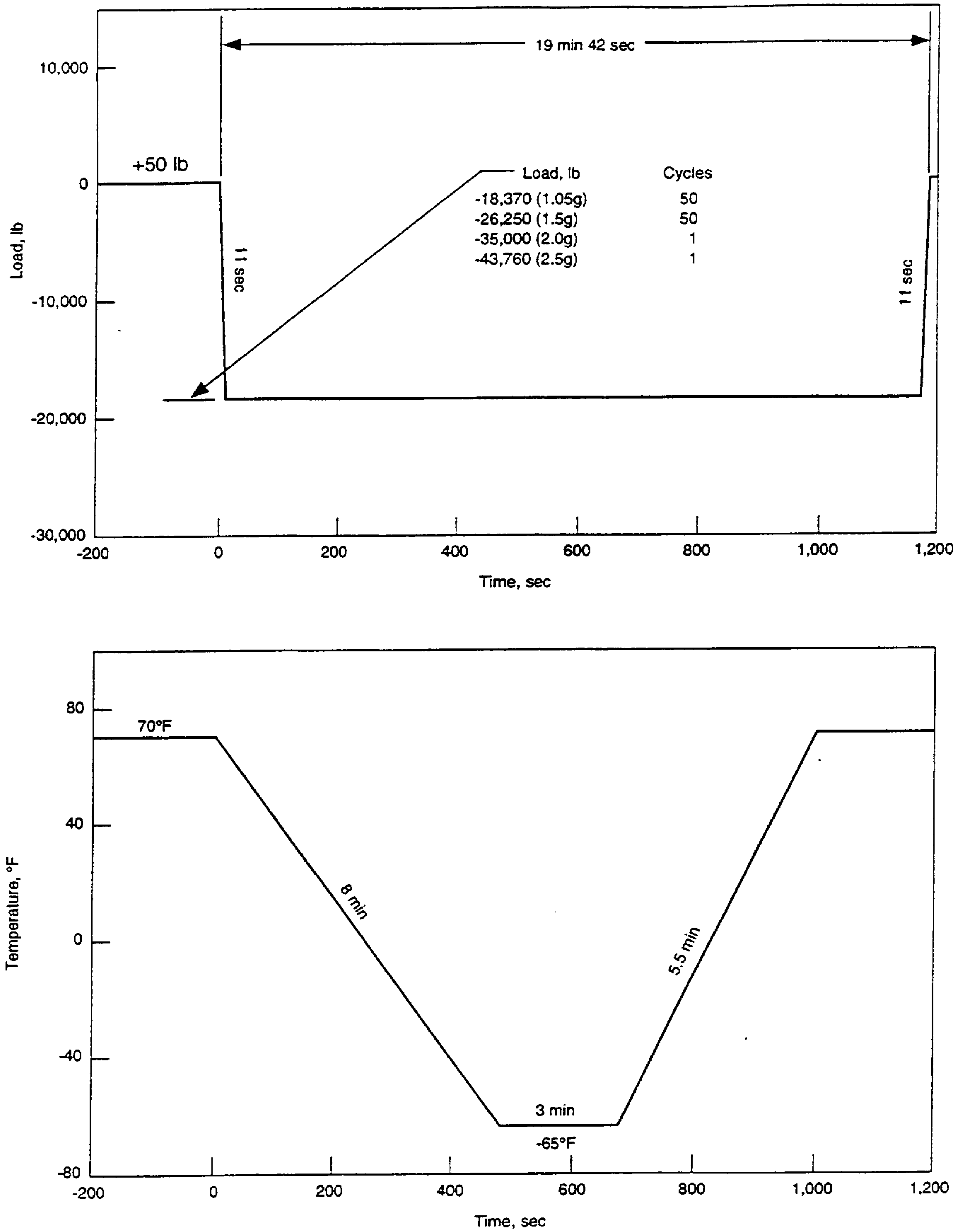

Figure B-3. Joint Thermal/Load Cycle

Joint compression testing under the 1.05g and 1.5g load cases did not cause any discernible damage, indicating that these systems are resistant to conditions expected during the HLFC flight experiment. Cracks started during the 2g loading and were propagated under the 2.5g loading.

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13. ABSTRACT (*Maximum 200 words*)

This document describes the design, fabrication, and installation of the suction panel and the required support structure, ducting, valving, and high-lift system (Krueger flaps) for flight demonstration of hybrid laminar flow control on the Boeing 757 airplane.

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